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A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 2007 to March 2009

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ABSTRACT

This document has been prepared for presentation to the 31st Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Rotterdam, the Netherlands, 25th and 26 May 2009. Brief summaries and references are provided on the aircraft fatigue research and associated activities of research laboratories, universities, and aerospace companies in Australia and New Zealand during the period April 2007 to March 2009. The review covers fatigue–related research programs as well as fatigue investigations on specific military and civil aircraft.

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A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 2007 to March 2009

Executive Summary

The Australasian delegate to the International Committee on Aeronautical Fatigue (ICAF) is responsible for preparing a review of aeronautical fatigue research in Australia and New Zealand for presentation at the biennial ICAF conference. The Defence Science and Technology Organisation (DSTO) supports the Australasian delegate to ICAF by publishing the review as a DSTO document. This document later forms a chapter of the ICAF conference minutes published by the conference host nation. The format of the review reflects ICAF requirements.

This review of research activities in the period April 2007 to March 2009 was undertaken by the Australasian National Delegate, Prof Graham Clark, of RMIT University, and Dr David Saunders, Research Leader Air Vehicles Division, DSTO.

Contents

8. Australasian Review of Aeronautical Fatigue Research

8.1 INTRODUCTION

This document presents a review of Australian and New Zealand work in fields relating to aeronautical fatigue in the period 2007 to 2009, and is made up from inputs from the organisations listed below. The authors acknowledge these contributions with appreciation. Enquiries should be addressed to the person identified against the item of interest.

DSTO	Defence Science and Technology Organisation, 506 Lorimer Street, Fishermans Bend VIC 3207, Australia
UNSW	University of New South Wales, Kensington, NSW, Australia
CASA	Civil Aviation Safety Authority, 16 Furzer St., Phillip, ACT 2606, Australia.
DTA	Defence Technology Agency, Auckland Naval Base, New Zealand
CRC-ACS	Cooperative Research Centre for Advanced Composite Structures. 506 Lorimer Street, Fishermans Bend, VIC 3207, Australia.
QinetiQ Aerostructures Australia	Level 3, 210 Kings Way, South Melbourne, VIC 3205, Australia.
DGTA	Director General Technical Airworthiness, RAAF Williams, Laverton Vic 3027, Australia
RAAF	Tactical Fighter Logistics Management Unit, RAAF Base Williamtown, NSW.
Boeing Australia Limited	363 Adelaide St, PO Box 767, Brisbane QLD 4001, Australia
RMIT University	Bundoora, VIC 3083

8.2 FATIGUE INVESTIGATIONS ON MILITARY AIRCRAFT

8.2.1 F/A-18 Flaw IdentificatioN through the Application of Loading (FINAL) Program *(G Swanton [DSTO])*

For the past several years, DSTO has been conducting the Flaw IdentificatioN through the Application of Loading (FINAL) program on ex-service "classic" F/A-18 wing attachment bulkheads, also known as centre barrels [1]. These ex-service centre barrels have become available to DSTO due to the Royal Australian Air Force (RAAF), US Navy and Canadian Forces embarking on a Centre Barrel Replacement (CBR) program for some their respective high time fleet aircraft. The RAAF CBR program is part of the Hornet UpGrade (HUG) Phase 3 program.

The FINAL program uses purpose built test rigs to apply Wing Root Bending Moment fatigue loads to each centre barrel and cycle them until failure. Fractured sections of the centre barrels are repaired to allow testing to continue and crack growth data on other centre barrel locations obtained (Figure 1). This is done in order to increase the amount of representative damage that can be detected using Non-Destructive Inspection during teardown. To date, 11 centre barrels have been tested to failure, which include test articles from ex-RAAF, US Navy and Canadian Forces aircraft. Quantitative Fractography (QF) has been conducted on the failure locations of the centre barrels cycled, as well as at other areas of interest for the maintenance of RAAF F/A-18 structural integrity. So far, some 80 QF reports have been completed.

As a cost reduction measure, the number of CBRs proposed for the RAAF fleet has been reduced significantly. The engineering rationale for this measure was largely achieved by using FINAL data to demonstrate that the incorporation times for planned structural modifications and CBRs could be extended to a point whereby aircraft would be withdrawn from service before they were required. Using crack growth rates from the International Follow On Structural Test Project (IFOSTP) centre fuselage fatigue test, the critical crack sizes determined from FINAL, and the observed exponential crack growth behaviour of cracks in F/A-18 7050-T7451 aluminium alloy structure, the Safe Life limits of many discrete locations could be extended [2]. In some cases, finite element modelling was also used to support the analyses and a DSTO-developed thermographic technique was used to verify the stress distribution fields (Figure 2). In all, this FINAL work is expected to result in projected savings to the Commonwealth in the order of $400 million [3]

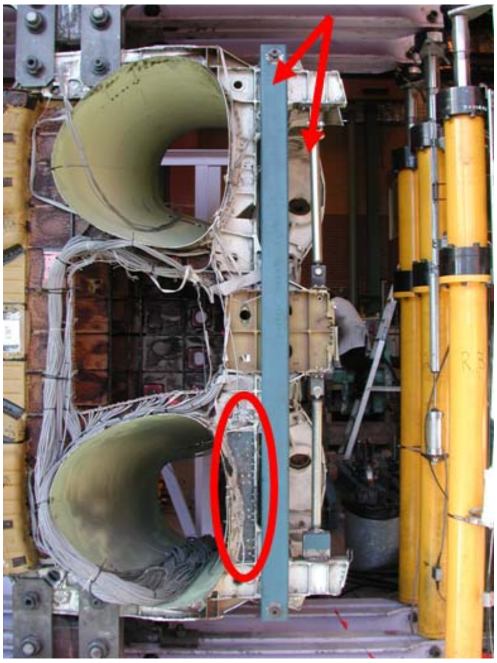

Figure 1 Repairs on a centre barrel test article. Strap repair (circled) and beam brace and turnbuckle modifications (arrows)

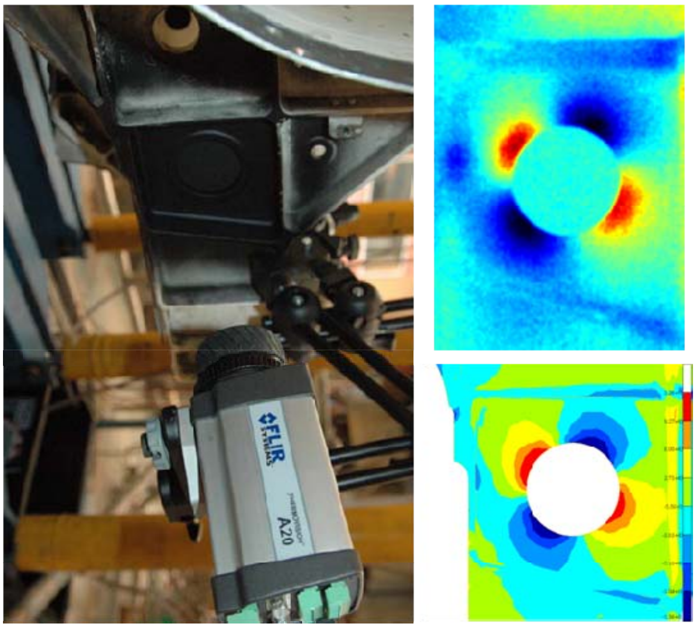

Figure 2 Thermal imaging technology was used to determine stress fields around structural details which was then used to validate finite element models.

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8.2.2 RAAF F/A-18 A/B Centre Barrel Life Extension (FINAL) *(J Moews, S Bandara, D Moorhead, B Gillot and T Nelson [QinetiQ AeroStructures])*

The Royal Australian Air Force (RAAF) operates 71 F/A-18 A/B aircraft. The F/A-18 A is operated in a fighter attack role and the F/B-18 B is a twin seat trainer. The F/A-18 A/B Hornet was designed to a safe life of 6000 flight hours under United States Navy (USN) usage. The Original Equipment Manufacturer (OEM) conducted many tests to validate this life. The International Follow-On Structural Test Project (IFOSTP) was a collaboration between the RAAF and the Canadian Forces (CF) to conduct further full scale testing of the F/A-18 A/B Hornet to verify the fatigue life for RAAF and CF configuration and usage. IFOSTP consisted of three Full Scale Fatigue Tests (FSFTs): FT46 for aft fuselage and empennage, FT55 for centre fuselage and FT245 for the wing. The IFOSTP tests were conducted using representative RAAF and CF manoeuvre and dynamic loads spectra. IFOSTP FSFTs are RAAF certification tests and underpin all fatigue management requirements of the Aircraft Structural Integrity Program (ASIP).

In addition to the IFOSTP tests, Defence Science and Technology Organisation (DSTO) conducted a component fatigue program for retired USN, CF and RAAF Centre Barrels (CBs) dubbed Flaw IdentificatioN through the Application of Loads (FINAL). The FINAL test program cycled high life ex-service F/A-18 A-D CBs under a Wing Root Bending Moment (WRBM) spectrum. The CBs were cycled to destruction so that the crack growth characteristics and critical flaw sizes at fatigue critical locations could be determined. In some instances, induced damage was required in order to produce failure, and where there was no prior failure, a Residual Strength Test (RST) was performed. Upon completion of testing, the CBs were torn down, examined using Non-Destructive Inspection (NDI) and the fracture surfaces analysed using Quantitative Fractography (QF).

The initial test interpretation of the IFOSTP data resulted in a set of fatigue management requirements that stipulated a major Structural Refurbishment Program (SRP) involving CB replacement. For flexibility in availability requirements and other reasons, Tactical Fighter Systems Program Office (TFSPO) required that the thresholds for SRP be extended. This requirement equated to a large amount of work re-assessing test interpretation data and assessing locations that were to be modified or replaced at the original SRP threshold limits. The DSTO FINAL test program was refocused on this TFSPO requirement and QinetiQ AeroStructures was tasked with carrying out project tracking and the analysis of critical locations not dependent on FINAL results.

The task confronting the combined DSTO/QinetiQ AeroStructures team was to extend the safe life of approximately 64 locations throughout the CB (as well as some wing locations) to the new set of limits. This was to be achieved within 12 months utilising existing IFOSTP data, FINAL testing and a set of specific F/A18-A/B structural assessment tools developed by DSTO and the RAAF Aircraft Structural Integrity – Directorate General Technical Airworthiness (ASI-DGTA).

Almost all goals of this ambitious project were met within the 12 month timeframe. All but a handful of the 64 locations were cleared to their revised target lives and the locations not cleared were generally easily inspectable and could be managed by TFSPO using Safety By Inspection (SBI). A couple of locations not cleared and not easily inspectable are the subject of ongoing work by both DSTO and QinetiQ AeroStructures.

8.2.3 Enhanced Teardown of Ex-service F/A-18A/B/C/D Centre Fuselages *(L Molent, S Barter, B Dixon [DSTO] and B Main [RAAF Directorate General Technical Airworthiness])*

Paper to be presented at ICAF 2009 Symposium

Abstract

The usage of the F/A-18 aircraft in service with Royal Australian Air Force (RAAF) was revealed to be significantly different to the design spectrum used by the United States Navy for structural certification. The differences lead to further fatigue testing in a joint Canadian and Australian program, using usage representative of both countries in the International Follow-On Structural Test Project (IFOSTP). These tests produced data sufficient to establish the structural integrity basis of both countries' aircraft. These consisted of the centre fuselage and wing tests (the responsibility of Canada) and an aft fuselage and empennage test performed by the RAAF.

Following the centre fuselage test, it was found that the aluminium 7050-T7451 centre barrel (CB) bulkhead's safe-lives were insufficient (without modification) to meet the required RAAF planned withdrawal date. Thus a CB-replacement (CBR) program was investigated. The Canadians and USN had already commenced a CBR program. For RAAF implementation two main problems were highlighted. The program would be difficult to run in-country and the availability of aircraft during the program would be insufficient to meet the operational needs of the RAAF. For these reasons, combined with the predicted expense of such a program, the RAAF examined alternative strategies.

Additionally, to mitigate against possible uncertainties in the CBR program that were not accounted for in the IFOSTP (eg. the potential onset of wide spread fatigue damage, additional failure locations, in-service induced defects etc) a teardown and inspection of several ex-service CB's was conducted by DSTO. To increase the probability of detecting existing "sub-critical" cracks accelerated fatigue testing of the CBs was conducted. This involved the application of representative cyclic loads to the retired CBs in a test rig. Loading was of sufficient magnitude and duration to ensure that potentially life-limiting cracks would be grown to a size that would ensure their detection. This program also allowed a re-examination of the earlier assumptions used to life the CB (eg. critical crack lengths etc). This resulted in a reduction of the number of CBRs required and thus significant savings of cost and increased aircraft availability for the RAAF.

To date, 12 CBs have been tested and torn down including two from RAAF aircraft Teardown inspection consisted of dismantling the CBs to the part level, followed by a detailed inspection of all areas of structural significance. In many cases, quantitative fractography (QF) was performed on the main cracks detected. Data from this program included the type and size of fatigue initiating defects, their locations and distribution, and fatigue crack curves that included in-service growth. It was found that the locations of final failures were very consistent between the CBs tested. This allowed an investigation of the fatigue scatter at discrete locations to be evaluated.

This paper briefly summaries the results of the accelerated fatigue testing program, including how these where used to reassess the life of the CBs and evaluate the scatter in fatigue for discrete locations.

8.2.4 The Effective Block Approach to Crack Growth Modelling (*M McDonald, .L Molent, and W Zhuang [DSTO]*)

Accurate fatigue prediction tools are essential for fatigue life management of fighter aircraft. Unfortunately conventional damage models (eg. AFGROW, Fastran etc) provided poor representations of crack growth data generated under typical RAAF fighter spectra [1], and this prompted the development of a predictive method tailored to variable amplitude data.

A key concept used is one where a repeating "block" of loading is applied throughout the fatigue life, sufficiently often that the blocks can be treated in a similar manner to single cycles in constant-amplitude loading. This characteristic approach to fatigue life prediction was first proposed by Paris [2]. The basic hypothesis is that the variations of the crack tip fields are describable in terms of some characteristic stress-related measure, for example, Root Mean Squared (RMS) of the stress intensity factor range (ΔK_{RMS}) (or the peak value).

Analysis of fatigue cracking over many decades has provided DSTO with a capability to produce detailed crack growth information for variable-amplitude loading under service conditions or laboratory conditions which closely resemble real service conditions. The methods being developed capitalize on the quality of this data and are aimed at providing useful tools for fatigue management. They have been reviewed in detail [3-9], and most recently applied to the life assessment of the wing of the RAAF's F-111C [8].

One simple formulation is to use the well-known Paris equation and apply it to variable amplitude loading and is known as the Effective Block Approach (EBA):

The first step in the process was to derive a crack growth resistance relationship based on the quantitative fractography (QF) coupon data for several fighter aircraft load sequences. To do this, the crack growth gradient (da/dB where B= spectrum blocks) and ΔK_{Peak} (=$\beta\ \Delta\sigma_{Peak}\ \sqrt{(\pi a)}$) were calculated for each consecutive QF data point. In this way the "average" crack growth *per block* is captured, thus accounting for any inherent sequence effects. In many instances the slope (equivalent to a Paris constant amplitude exponent of m) was found to be approximately 2 and the intercept (C) varied with the spectrum and stress level (with material and geometry constant). A Paris exponent of approximately two (ie. exponential growth) has been found to hold for a large range of variable amplitude spectra and materials [4, 11]. An example of a prediction is shown in Figure 3.

A key outcome from this analysis is that by determining C and m, it appears to be possible to use these parameters, determined using one particular fighter sequence, to predict behaviour under a different sequence. This involves coupon-derived data for just one sequence, and the use of conventional fatigue prediction models – which can be fairly inaccurate in direct prediction – simply as a transfer tool [3-9]. Independent review of the methodology continues, eg. [12].

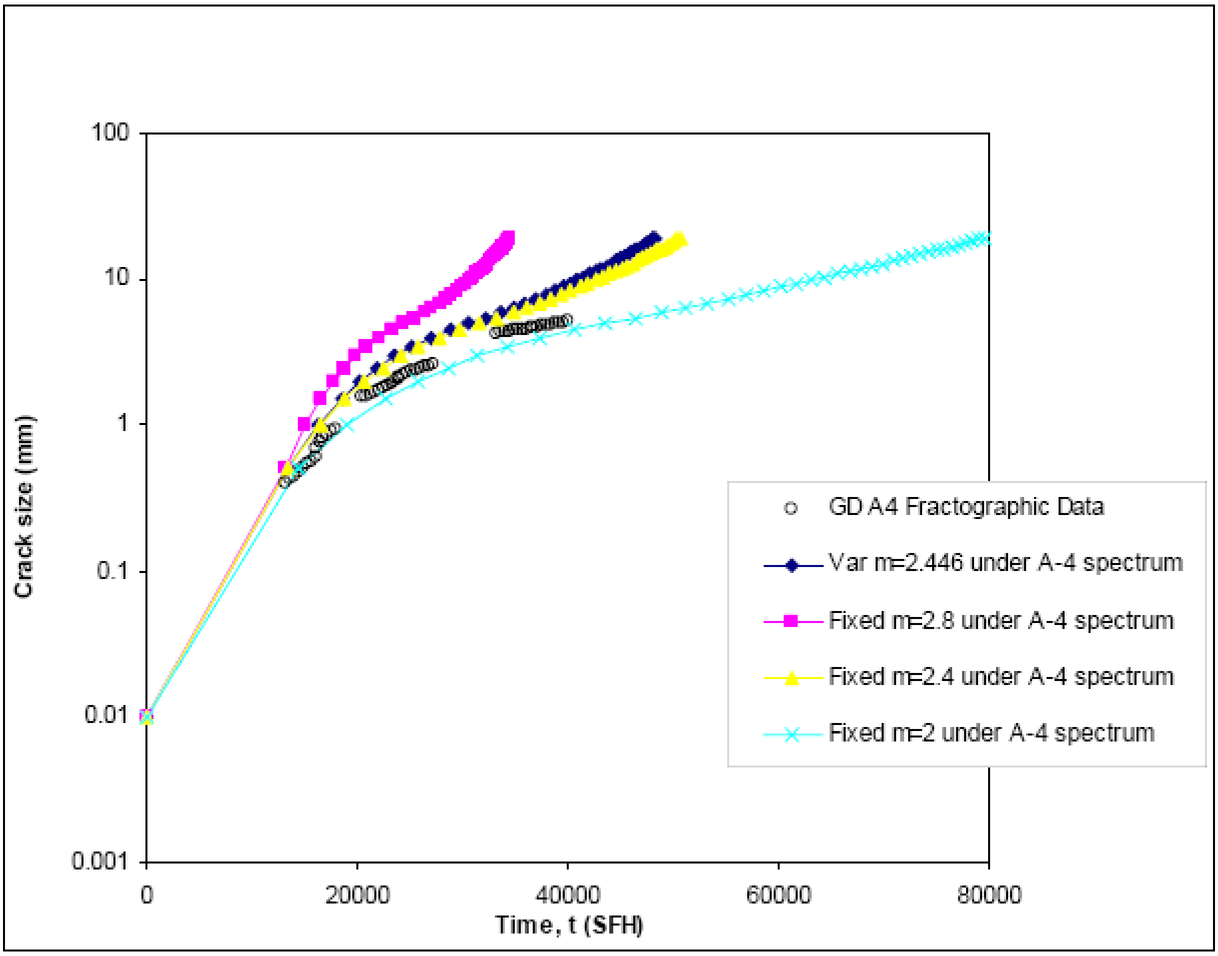

Figure 3 Prediction of the early General dynamics (GD) F-111 D6ac steel Wing Pivot Fitting Splice failure [10]. The affect of varying the Paris-like m parameter is shown.

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8.2.5 F-111 Wing Life Validation from Full-Scale Testing *(R Boykett, [DSTO])*

After the United States Air Force (USAF) retired their fleet of F-111 aircraft in 1998, Australia became the only operator of the F-111 in the world. A partnership between the Royal Australian Air Force (RAAF), the Defence Science and Technology Organisation (DSTO) and Australian Industry created the "F-111 Sole Operator Program (SOP)". This successfully ensured that support for the aircraft is provided in the absence of the Original Equipment Manufacturer (OEM) of the aircraft, General Dynamics (now Lockheed Martin).

In one important example of the SOP, the outcome of DSTO's C-model wing testing ultimately led to their retirement from RAAF service (completed in 2006) and the need to provide a certification basis for the replacement D and F-model wings. This led to the testing of an ex-USAF F-111F short wing termed the F-111F Wing Economic Life Determination (F-WELD) test [1]. This work enabled an assessment of all critical and significant areas to effectively support the RAAF's structural integrity management of F-111 wings until the fleet's Planned Withdrawal Date (PWD). It also provided an opportunity to develop Non-Destructive Test (NDT) procedures using the Science Applications International Corporation (SAIC) Ultra Image International Ultraspect-MP automated ultrasonic NDT scanning system [2-4].

The cyclic test loading (2004-2007) on the F-WELD test wing, accrued 37,888 simulated flight hours before being terminated for economical reasons. The completion of the subsequent teardown activity (*Figure 4)* preceded quantitative fractography (QF) and complementary analyses in order to compile a full suite of data from which to support structural integrity management actions. This work assessed all the defect indications identified on the test article, using results from Lower Wing Skin (LWS) inspections, build quality (BQ) assessments [5], detailed crack investigations, and fatigue lifing results. QF analyses were conducted to generate crack growth data (e.g. *Figure 5*) for numerous Taper-lok fastener holes in both the LWS and spar between the wing splice and forward auxiliary spar station, FASS 226 as well as cracking in the Inboard Pivot Pylon (IPP) housing. This revealed cracking in only 1.9% of the 1,487 Taper-lok fastener holes inspected. In addition, wide spread fatigue cracking was not observed and the discovery of corrosion was minimal.

NDI before and after dismantling revealed cracking in five sealant injection holes (SIHs) in the LWS, a feature that has not been considered or highlighted in previous wing tests and for which there was no known precedence in service. They have now been added to the list of control points for the wing.

The information generated now assists the RAAF to make informed decisions about management strategies for F-111 wings until PWD.

Figure 4 a) Upper Wing Skin removal; b) Separation of the Wing Pivot Fitting (WPF) and Inboard Pivot Pylon (IPP); c)WPF Lower Plate; d) Lower Wing Skin (LWS) Hardpoints; e) IPP Housing; f) Spar fragment; g) Example of water picked fragments; h) Fragment prior to being broken open for fractographic analysis.

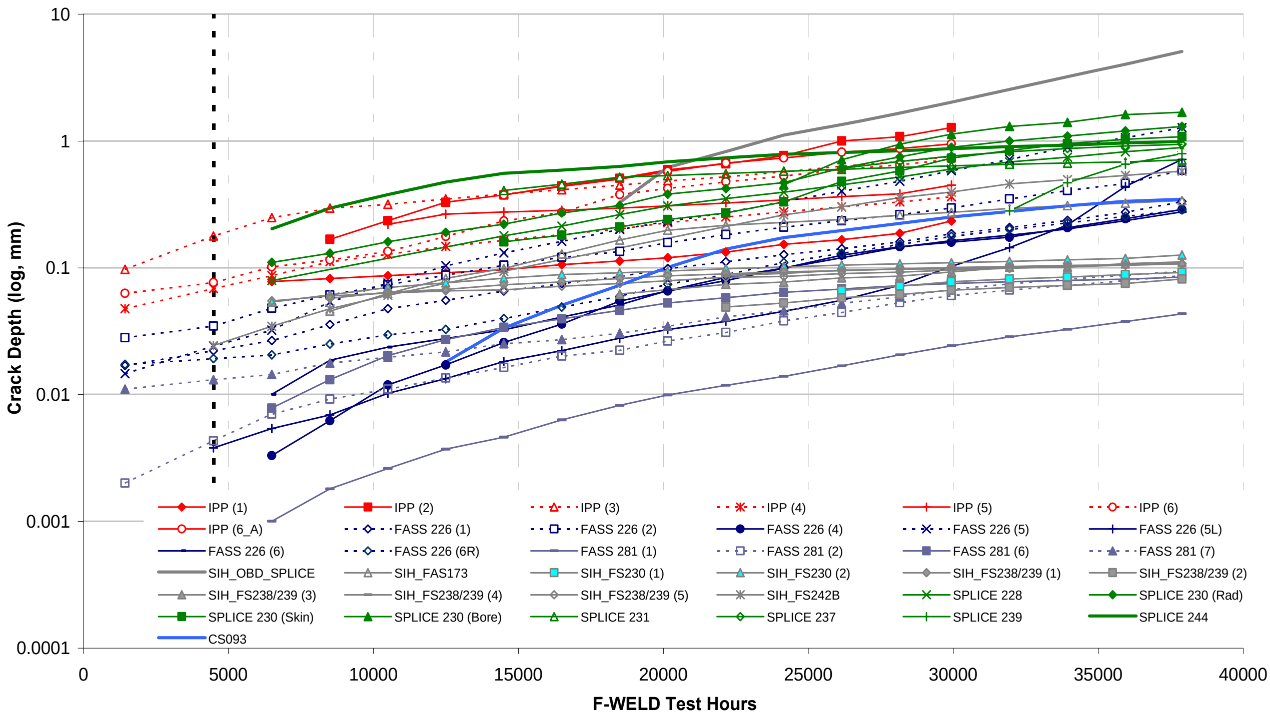

Figure 5   Examples of crack growth curves from the LWS of F-WELD Test Article

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8.2.6 Fatigue Life Analysis for F-111 FS 496 Nacelle Former and the New Methods Developed for Structural Integrity Analysis, *(G Chen [DSTO])*

The fuselage station 496 (FS 496) nacelle former is a high strength D6ac welded steel structure situated near the entrance to each engine air inlet and provides support to the forward inlet structure. Cracking of the nacelle formers has been a problem for both the Royal Australian Air Force (RAAF) and the United States Air Force (USAF). FS 496 presents a significant and complicated problem, as the combination of loading and structural details cause cracking. Typically, a long life prediction was given from OEM: 50,000 flight hours for DI 25G, leading to an inspection interval of 25,000 hours. Although twenty years efforts from OEM and others were devoted to solving this problem, there has not been any better correlation between the results and in service findings to date. The analytical results of the cracking to date from Lockheed Martin (e.g. 25,000 flight hours) vary significantly from the current fleet management data (e.g. 525 flight hours) for the critical locations.

Extensive studies were carried out in DSTO considering the critical front intake panels of FS 496 and the potentially significant air intake pressure loads. These ranged from intake structural details investigation; analytical and computational fluid analyses; load and usage investigations; and risk, propulsion, stress, regression and crack growth analyses.

New methods were developed to deal with analytical issues as they arose. These new approaches include; a pseudo-compressible theory for air-involved fluid mechanics, an Iterative Air Flow approach to achieve the convergent solution for the three-dimensional Computational Fluid Dynamic Analysis, a pseudo-compressible approach for propulsion analysis of the intake to enable limited load cases to be selected from unlimited data, a new envelope method for stress analysis, a Stage Policy method for regression analysis, and finally at the end the focus on notch plasticity in the crack growth analysis. The work is summarised as the following:

A cracking history review was carried out to gain an understanding of the cracking related issues for the F-111 FS 496 nacelle former. A statistical approach to the RAAF service history predicted a Safe Life estimate of 2,270 flight hours for the DI 25 region with 95% confidence and 1 in 2000 risk based on a failure conservatively defined as the occurrence of a crack of any size.

Theoretical fluid methods were then explored to obtain intake loads and pressure loads information. Three analytical approaches were developed. Approach I, the conventional incompressible theory without considering changes of density, has explicit-form solutions, thus is simple and easy to implement, but is least accurate. Approach II, using conventional compressible theory, is closer to the reality of airflow problems by considering the changes of density, thus is more accurate but has no explicit-form solutions and thus is relatively more complicated and difficult to apply. A new pseudo-compressible approach, Approach III, was developed in this report to consider the change of density to produce reasonably accurate results, with the advantages of explicit-form solutions, which are easy to implement. The predictions from Approach III are closer to those of the more complex conventional compressible theory of Approach II compared to those of Approach I. The results of the pressures provide boundary conditions for a three-dimensional CFD model of the F-111 intake. For the cases studied here, Approach III is recommended.

A three-dimensional computational fluid model was created for the engine intake of the F-111. The boundary conditions were initially derived from a simplified pseudo-compressible fluid flow approach, and then were updated through iterative processing. The three-dimensional CFD analyses were carried out for a range of free stream Mach numbers with a Reynolds-average approach. In these analyses, the standard k-ε model is used for its robustness, economy, and accuracy. Using the standard wall function for the near wall treatment substantially saves computational resources. The first-order upwind discretisation schemes were used in initial iterations for saving time, and the second-order upwind discretisation schemes were used in the final results for high accuracy. The CFD results not only provide important load information for the subsequent stress analyses and fatigue analyses, but also were used to validate and update a simplified envelope method.

In order to select limited cases to study in this work from the enormous number of possible flight conditions, a propulsion analysis was carried out, and an Intake Loads Program (ILP) for the F-111 engine model was developed based on the pseudo-compressible approach. From this program, the pressure loads along the intake can be obtained for any flight condition. A sensitivity analysis was also carried out. From ILP and the sensitivity study, the 15 flight conditions across the flight envelope were selected to be studied in this work from the range: Mach number from 0 to 2, altitude from 0 to 60,000ft and power setting from 1 to 6.

Using Intake Loads Program, in which the pseudo-compressible approach (Approach III) was used, the predicted pressures were compared with the measured flight data from (Evans, 1971), showing an excellent agreement. The difference is between 3.03% and 0.25% with an average being 1.29% for the 14 compared cases.

The original ILM FE model developed by Lockheed Martin was examined and modified by adding the additional intake structure. A stress analysis was carried out on the F-111 FS 496 former and surrounding intake structure. There is a significant difference in the stress results at FS 496 between the updated and original ILM FE model. A sensitivity study was subsequently carried out to compare a real pressure distribution and different functions representing the pressure distribution. It was concluded that the results were not greatly affected by the different functions. In other cases investigated, when the distributions of air pressure along the intake wall were not known, an "envelope" approach was developed for the stress analysis of FS 496. This approach assumes that the distributions of the air pressures along the wall of the intake structure are within an envelope. The envelope is defined based on three fluid dynamic theory approaches: conventional incompressible flow, conventional compressible flow, and the new pseudo-compressible flow

approach. The pseudo-compressible approach produced similar predictions to the more complex conventional compressible approach. For solving similar engineering problems, the pseudo-compressible approach is recommended.

The results from the stress analysis of updated ILM were used in a Fine Grid (FG) FE model for FS 496 and a more detailed stress analysis was carried out. The Freebody loads were extracted from ILM, and were distributed onto the grid points of the FG FE model. The intake load cases were separated into two main categories, dependent on loading, for the analysis of the critical locations. The first category was positive pressure acting on the intake internal panel. The maximum and minimum principal stress distributions around the critical location were similar among all intake load cases within each group, varying mainly in magnitude. The second category was negative pressure loads acting on the intake panel and included all remaining load cases. The maximum and minimum principal stress distributions around DI 25 were similar among all intake load cases within each group, varying mainly in magnitude. FE results for FG FS 496 model were obtained.

Finally, a numerical procedure was developed for the modelling of fatigue crack growth for the critical location where the intake loads were considered. The stress distribution in the vicinity of the notch was calculated based on a non-linear, kinematic hardening, cyclic plasticity model and the generalized Neuber's rule. Numerical results showed that the effect of intake loads may come from two aspects. The intake loads not only have an impact on the magnitude of the local stress but may also shift the mean load of the subsequent load spectrum down. The overall effect retards the fatigue crack growth. The resulting prediction with consideration of the intake effect correlates better with in-service cracking data than the prediction for crack growth without consideration of the intake loads.

A fatigue life prediction for the critical location was produced, which correlated extremely well with the in-service failure data and showed that the existing management strategy of an inspection interval was conservative.

The outcomes from this work provide the RAAF with the necessary advice to safely manage the F-111 fleet to the Planned Withdrawal Date (PWD) at an important location in the fuselage. The methods developed in the work are applicable for similar engineering problem**.**

8.2.7 F-111 Individual Aircraft Tracking System *(K Jackson, C Stephens and R Cave [QinetiQ AeroStructures])*

QinetiQ AeroStructures were recently engaged in a two-year project for the RAAF, the F-111 Aircraft Structural Integrity Program (ASIP) Consolidation Project (F-ACP) which brought to maturity key elements of the F-111 ASIP using tools and data developed as part of the F-111 Sole Operator Program (SOP). One of the elements of the F-ACP was the development of an Individual Aircraft Tracking (IAT) system to assess the usage severity of individual aircraft relative to the fleet average usage spectrum (DADTA3).

Investigation into available options (Nz, NzW, segment and mission profiling) led to the development of an IAT software tool which creates a spectrum based on individual aircraft NzW exceedance counts and uses this to perform a Linear Elastic Fracture Mechanics (LEFM) analysis at a representative location.

To test the IAT system the parametric flight data (which is available on only a few aircraft) used to create the DADTA3 spectrum was divided into three separate test spectra. The NzW exceedance data associated with each test spectra was run through the IAT system and a severity factor relative to the DADTA3 baseline case established. The severity factor was compared with the actual severity factor determined using LEFM at a range of fatigue critical locations on the aircraft. The testing showed that the IAT severity factor was sufficiently indicative of usage severity over the majority of the airframe to warrant its use for overall usage severity assessment. The key structural regions not covered by the IAT system are compression dominated locations and the horizontal tail.

The IAT system is currently being implemented and is to be used in the following ways:

a. The usage severity factor based on individual aircraft operations since each of the previous major servicings (inspections) will be incorporated into quarterly usage reports. By reporting usage severity since the previous inspections the effect of the usage on the Safety By Inspection (SBI) program can be determined. It is expected that the routine reporting of severity data will assist with fleet levelling.

b. The fatigue management of the wing lower skin was recently transitioned to safe life based on full scale fatigue tests performed by DSTO. By using the IAT severity factor for all time RAAF usage when compared to fatigue test spectra a more accurate safe life limit can be determined.

c. In assessing Maintenance Interval Extension Requests (MIERs) a quantitative measure of the severity of the previous usage allows a more accurate determination on the acceptability to be made.

8.2.8 Structural Health Monitoring of a Bonded Composite Patch Repair on a Fatigue-Cracked F-111C Wing *(A Baker and Nik Rajic [DSTO])*

A large fatigue crack in the wing skin of an Australian Defence Force F-111C aircraft was repaired with an adhesively bonded boron/epoxy fibre composite patch, see *Figure 6*. The patch prevented further growth of the crack for 670 flying hours.

The aim of the F-111 Sole Operator Program is to ensure the continued safe operation of ADF F-111 aircraft out to its planned retirement. As part of this program, a decision was made in 1997 to assess the residual life of the F-111 wings in part by undertaking a fatigue test at DSTO on a retired wing. An additional aim of the test was to substantiate further the bonded repair, in view of its potential application to other F-111 aircraft – either as a repair for cracked wings or as reinforcement to inhibit fatigue crack formation in uncracked wings - since at this time (the late 1990s) the indications were that the Planned Withdrawal Date (PWD) for F-111 would possibly be in the 2012-2015 timeframe. The test importantly would also evaluate the effectiveness of non-destructive inspection (NDI) and other non-destructive procedures to detect crack growth under the patch or disbonding of the patch system. The repaired wing was therefore selected and used for this study. In this test the patch was successful in preventing growth of the crack for around a further 9000 simulated flying hours.

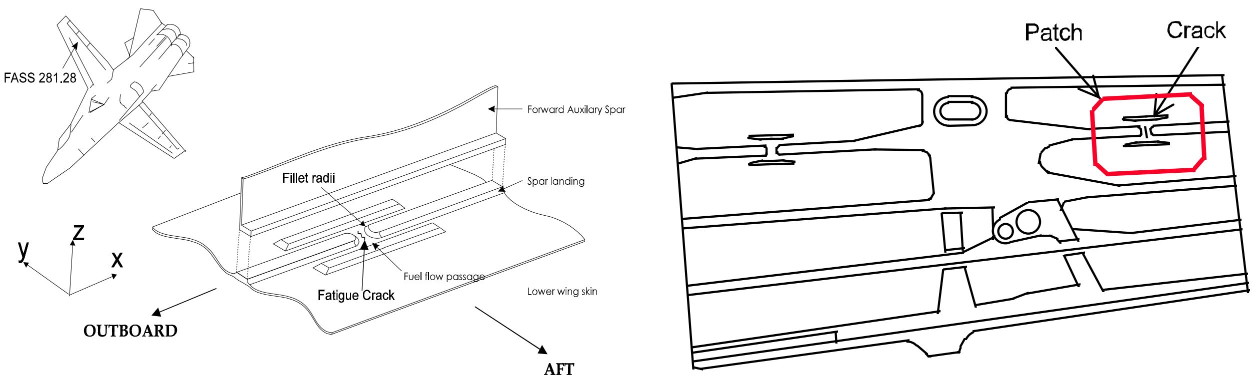

Figure 6 Location of cracking on the F-111 lower wing-skin, at FASS 281., and (right) Location of cracking on the F-111 lower wing-skin, at FASS 281, showing the outline of the repair patch on the outside surface, which is made of 14 ply boron/epoxy (0_2, ± 45, 0_3)$_S$ bonded with adhesive FM73, size 500 mm spanwise × 350 mm chordwise.

Figure 7 provides details of the locations of strain gauges in relation to the LWS repair, and *Figure 8* shows the boron/epoxy patch and strain gauges.

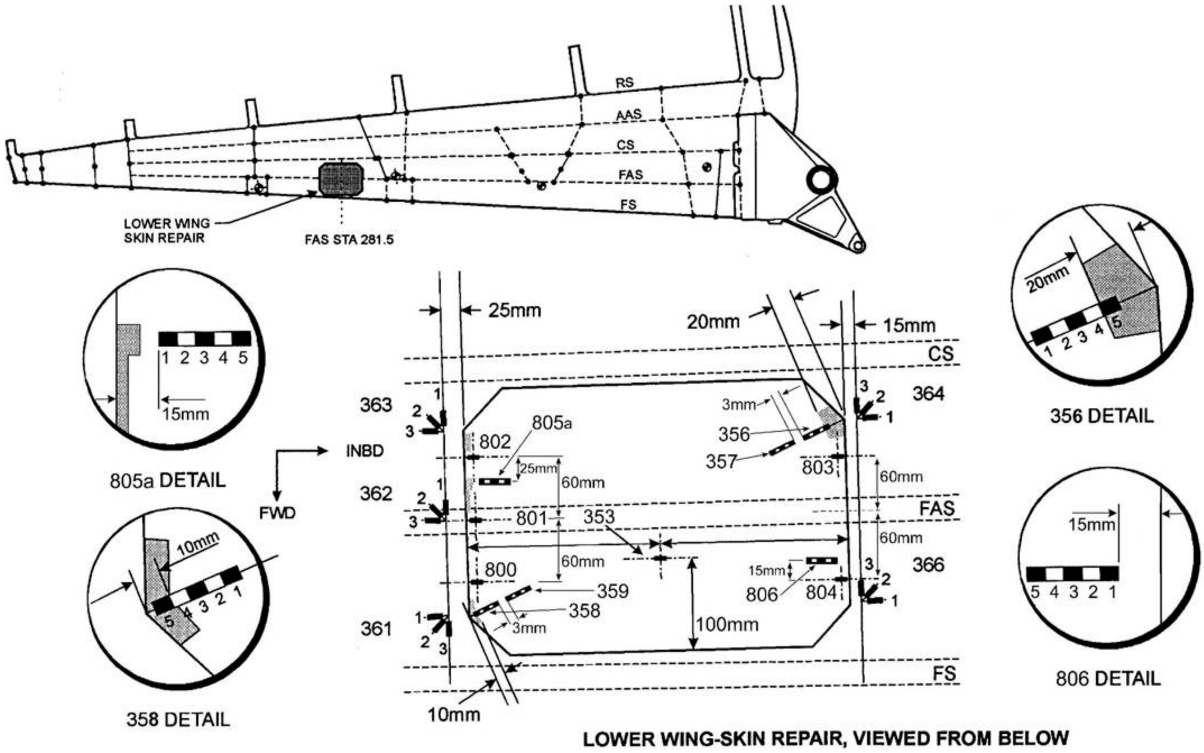

Figure 7 Location of strain gauges used on the patch and parent structure from block 12, also showing, schematically the position of some of the disbonds detected by NDI. Please note that prior to block 12 strain gauge, 354 was in the approximate position of 356 and 355 was in the approximate position of 358.

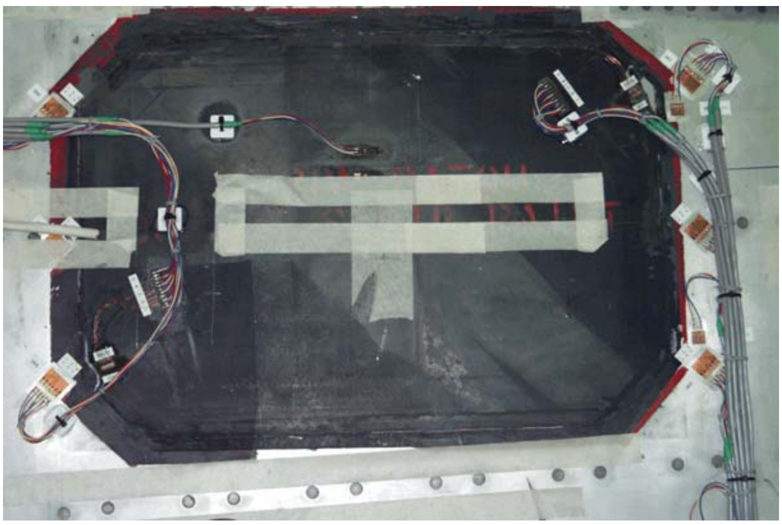

Figure 8 Boron/epoxy patch bonded to F-111 wing skin showing also some of the strain gauges used for SHM during the fatigue test to check for patch disbanding

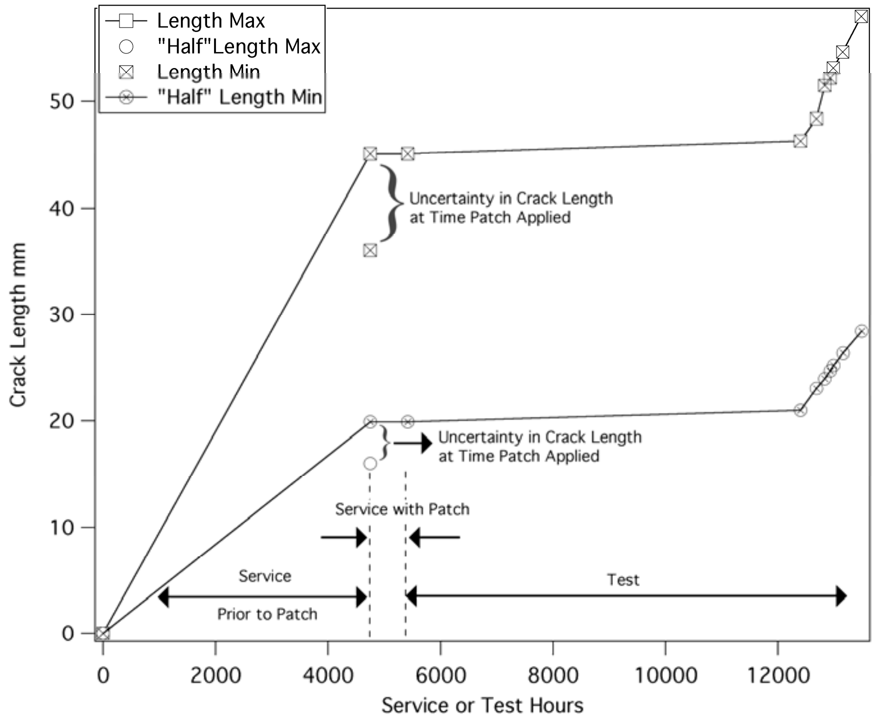

Figure 9 Plot of crack size over the life of the wing measured from fractographic analysis of the patched crack region

As a consequence of NDI indications of local patch disbonding and concern that it could lead to rapid growth of the repaired crack and thus premature failure of the wing a structural health monitoring (SHM) technique was implemented to monitor the patch.

A report has been prepared providing background to the repair and its performance, and describes a) the comparison of NDI results for crack detection with the fractographic studies on the crack after the tear down and b) the comparison of the NDI and SHM results for disbonding with bond-strength measurements on the patch system undertaken at the completion of the fatigue test [1].

Both NDI using on an ultrasonic technique and the SHM using a strain-based technique were successful in detecting disbonding of the patch. Based on the use of the bond-strength test as a destructive validation procedure it was concluded that both the NDI and the SHM techniques were highly effective in detecting disbonds and that the strain-based SHM technique (with several significant improvements) could be used for in-flight monitoring of repair patches in demanding or critical applications.

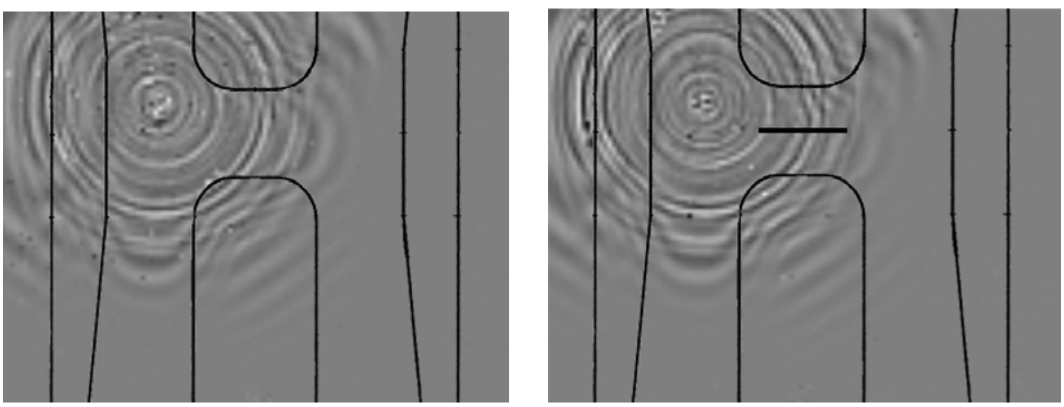

Figure 10 Elastic wave propagation a) in the absence and b) in the presence of the crack in a structural detail simulating the repaired region in the F111C wing.

More recently, studies on alternative SHM techniques, incorporating piezoelectric elements, on bonded repairs have been undertaken [2]. These studies have used the F-111 FASS 281 bonded repair problem as a vehicle to develop and

demonstrate a broad-field SHM technology using acoustic waves. The FASS281 case was considered an ideal development platform despite the imminent retirement of the F-111 (2010) as the problem involves most of the key issues affecting that viability of this form of SHM, including geometric complexity, crack closure, high operational loading, adverse environmental conditions and sensor embedment. The primary aim of this work is to develop a repair patch containing integrated piezoelectric elements.

It was planned to develop the diagnostic functionality of the patch in two steps: starting with the crack detection problem, followed by the patch system integrity problem. This staged approach was deemed essential because of limited resources and the need to concurrently address several key engineering issues that, if not resolved, would confine the technology to the laboratory bench, irrespective of its potential diagnostic benefit. Chief amongst these is transducer durability. At candidate sensor locations in the repaired F-111C wing structure, tensile strains of approximately 2200 microstrain are expected at the design limit load. Sustained strain levels of this order have been reported to produce deterioration in the performance of piezoceramic materials. However, our study has shown that the piezoceramic transducer elements may have the requisite durability for this application. These elements comprise a piezoceramic disc attached to a polymer film containing an electrical interconnect layer. A preliminary test of transducer durability under fatigue was made on a metal coupon exposed to cyclic loading at constant amplitude. Measurements of the transducer impedance-magnitude spectrum were taken throughout the fatigue test with *Figure 9* showing measurements at three key moments: (i) prior to commencement of loading, (ii) approximately mid-way through the test and (iii) at termination, caused by failure of the coupon after 24.8 million loading cycles. The comparison demonstrates stable electromechanical performance, which is a significant result given that tensile strain amplitudes were in the range of (2000 to 2500 microstrain) for approximately 10^6 load cycles.

Because of the high level of structural detail in the repair zone, initial work has focused on locating optimal positions for the placement of transducer elements so as to maximise the production of and sensitivity to interaction between the wave field and a crack. High resolution laser vibrometry was applied to map the wave-field in the F-111 FASS 281 repair zone. Scans were produced for the baseline structure and for a case involving a semi-elliptical notch 20 mm long, 1.8 mm deep and 1 mm wide located in the stiffener depression to simulate a crack. These are shown in *Figure 10*. The scans clearly convey the large influence that structural detail has on the elastic wave propagation through the coupon. By comparison the influence of the notch is only marginal, but nevertheless is noticeable as a slight attenuation of the field to the right of the notch. This shows strong scattering along a diagonal from the right hand side of the notch, which in this example represents a region of optimal sensor placement.

References
[1] Baker AA, .Structural Health Monitoring of a Bonded Composite Patch Repair on a Fatigue-Cracked F-111 Wing, DSTO-RR-0335.
[2] Baker A, Rajic N and Davis, C, Towards a practical structural health monitoring technology for patched cracks Composites: Part A (2008).

8.2.9 P-3C Service Life Assessment Program. *(L Meadows, K Walker, E Matricciani, K Maxfield, [DSTO] and J Duthie, [QinetiQ AeroStructures]).*

On behalf of the Royal Australian Air Force (RAAF), DSTO participated in the United States Navy (USN) P-3C Orion Service Life Assessment Program (SLAP) that ran from 1999 to 2006. The P-3C SLAP was a program of full-scale fatigue testing and analysis with the aim of providing structural clearance and service life extension for the P-3C fleets being flown by the respective defence forces. DSTO was tasked by the RAAF to produce a test interpretation report which translated the results from severe USN usage to average RAAF usage. The DSTO P-3C SLAP test interpretation was conducted to the RAAF requirements that amended the Certification Structural Design Standard by substituting the original Orion CAR 4b requirements with FAR 25.571 Amendment 25-95 for fatigue management of the aircraft via a Safety-By-Inspection (SBI) program. The fatigue life (crack initiation), crack growth life and total life (crack initiation plus crack growth) were determined by analysis, supported by coupon and full-scale fatigue test results for each Fatigue Critical Area (FCA). The crack initiation analyses were based on local notch strain linear cumulative damage analysis pegged to full-scale fatigue test results. The crack growth analyses were performed using the most advanced and accurate methods available, specifically the well known FASTRAN program which is based on an analytical plasticity induced crack closure model. The results of these analyses were published in late 2006.

<u>P3 Structural Management Plan.</u>

The raw results stemming from the P3 SLAP Test Interpretation process indicated that the calculated inspection thresholds had already been exceeded by the majority of the RAAF P3 fleet. A program was then undertaken to utilise the findings of fleet inspections in accordance with the guidance of FAA AC 25-21 in order to determine if a valid case existed to adjust the initial inspection thresholds. A probabilistic approach utilising actual crack findings from the USN fleet (which is well ahead of the RAAF fleet in terms of accumulated flying hours) was devised and implemented. This enabled a significant number of inspections to be deferred, however a small number were still required to be undertaken on a priority basis. These results then served as the basis for DSTO recommendations to the RAAF regarding structural inspections, modifications, replacements and redesigns to extend the operational life of the RAAF P-3C to their Planned Withdrawal Date in the form of a document called the Structural Management Plan (SMP). This was an extensive body of work undertaken by a large team from DSTO, their industry partner QinetiQ (formerly Aerostructures Australia) and the RAAF Directorate General Technical Airworthiness.

P3 Individual Aircraft Tracking.
DSTO was requested to develop an Individual Aircraft Tracking (IAT) program to track the fatigue life of the nineteen RAAF P-3C aircraft still in service in comparison with the damage induced on the P3SLAP Fatigue Test. The IAT system enables the calculation of accumulated fatigue damage at six tracking locations (see *Figure 11*). One of the key inputs is the load/stress at each location, and this is currently determined through a parametric based approach. The accuracy of this approach will be reviewed when data becomes available from six fleet aircraft which have been fitted with an Operational Loads Monitoring System (OLMS) which includes the installation of six strain sensors at a range of locations on the airframe (see *Figure 12*). DSTO developed a range of options for calibration of the OLMS system, the best of which is a full ground based calibration which is scheduled to commence from early 2010.

The SMP and IAT programs represent the end products for the RAAF from the SLAP, and will be used for the continual fatigue management of the RAAF P-3C fleet. The RAAF commenced the SBI program on the first 2 fleet aircraft in mid 2008. To reduce the economic impact of the SBI program due to the current frequency of re-occurring inspections, DSTO embarked on an extensive coupon and component test program aimed at increasing the SMP inspection intervals by removing conservatisms in the crack growth analysis tool used during the DSTO TI and SMP development process. An example of part of this program is explained in greater detail in the following synopsis on the crack growth tool development. The results from these test programs, and several other related activities, will feed in to a comprehensive review of the SMP scheduled to take place in 2010. The aim of the review is to reduce the conservatism in the current system such that all current 15,000 hour thresholds will be extended to 16,000 hours, and all inspection intervals will be extended to at least 2,500 hours.

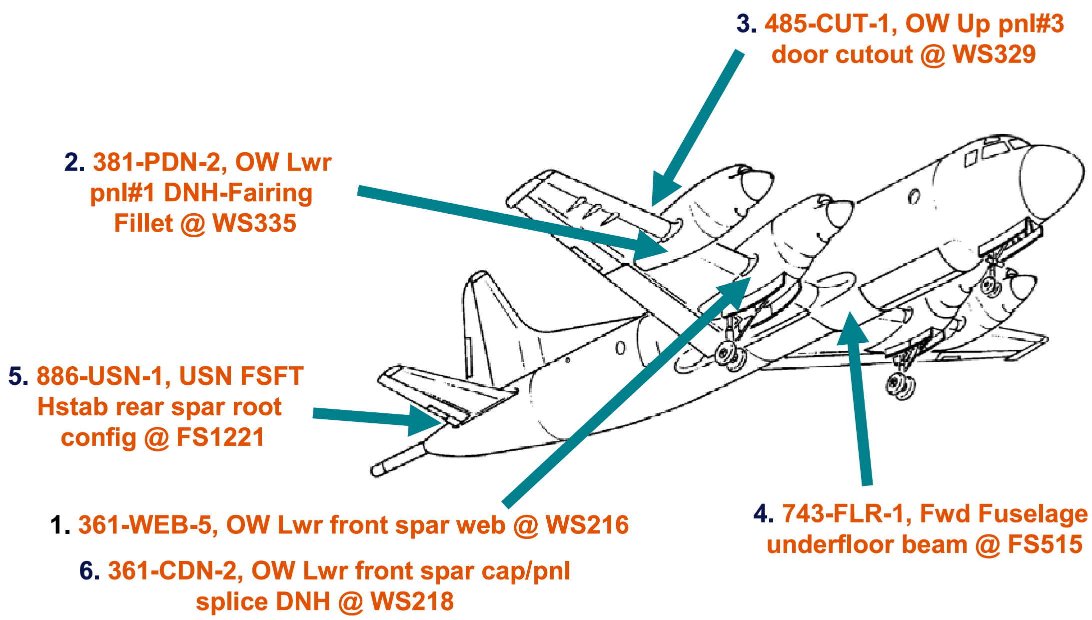

Figure 11 RAAF P-3C Orion IAT Tracking Locations

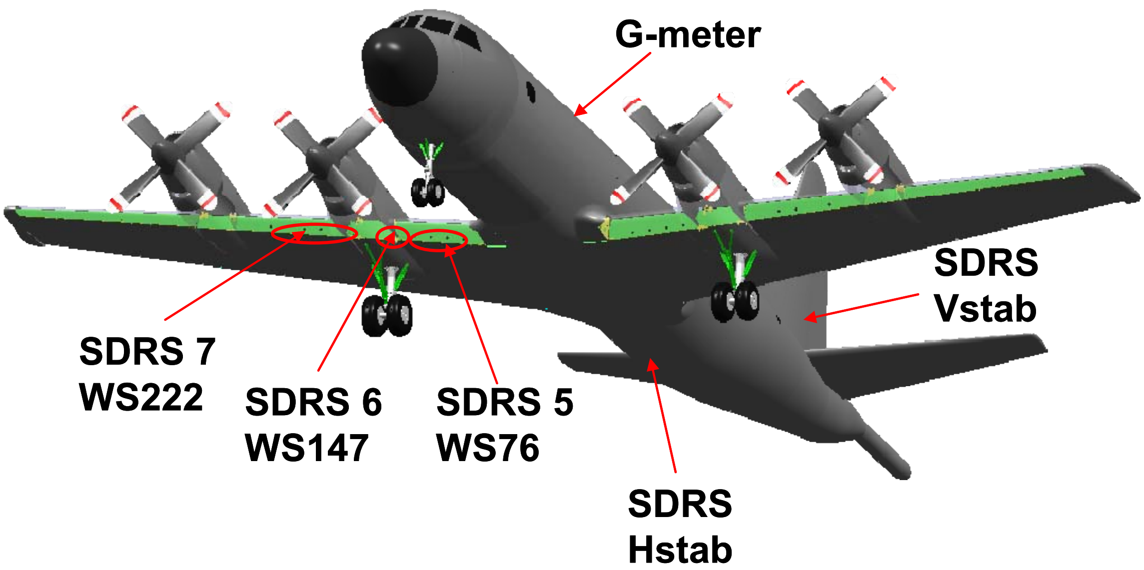

Figure 12: RAAF P-3C Orion OLMS Strain Gauge Locations

8.2.10 Development and validation of fatigue crack growth analysis tools and methods in support of P-3C structural integrity *(K Walker, E Matricciani, [DSTO] and J Duthie, [QinetiQ Aerostructures])*

Fatigue crack growth analyses in support of structural integrity management of the Royal Australian Air Force P-3C maritime surveillance aircraft fleet are performed using FASTRAN3.8, which uses a plasticity-induced analytical crack closure model to account for stress ratio and load sequence (retardation/acceleration) effects. Previous comparisons with coupon and component tests show that the analytical crack growth and inspection intervals based on FASTRAN analyses are often overly conservative under the P-3C spectra. Meanwhile, recent work by the original equipment manufacturer of the P-3C, Lockheed Martin, uncovered some errors and shortcomings in the methods used in the P-3C service life assessment program in determining the sequence of loads and stresses at certain aircraft locations. Considering that the previous coupon and full-scale tests were performed with load/stress sequences determined under the old Phase IIB load system, it was decided necessary to conduct coupon and component tests under the updated and more accurate Phase IIC sequences to reflect their effects on fatigue damage accumulation. This provided an opportunity to significantly improve the accuracy of FASTRAN analyses by reliably reducing the conservatism, thereby providing support to extend inspection intervals, and realise significant financial savings and improved aircraft availability. Consequently, coupon tests are being conducted to provide data for the re-calibration of FASTRAN. Tests are also being carried out to quantify total life from natural crack formation and cold working effects, and on sub-components representing complex structures. Results so far indicate that the modelling process can be improved as demonstrated in *Figure 13*. Examples of the component tests (in this case the wing panel splice) are shown in *Figure 14* and *Figure 15*.

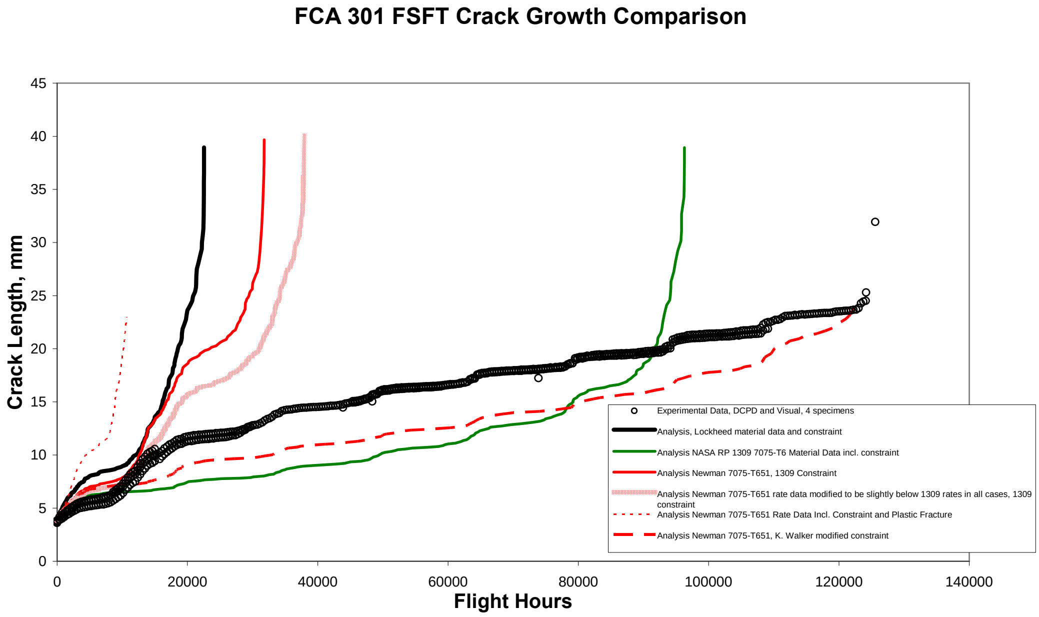

Figure 13 Comparison of experimental crack growth against FASTRAN analyses using various material data and constraint factors for FCA 301 FSFT. The red long dash curve indicates the improvement which may be possible.

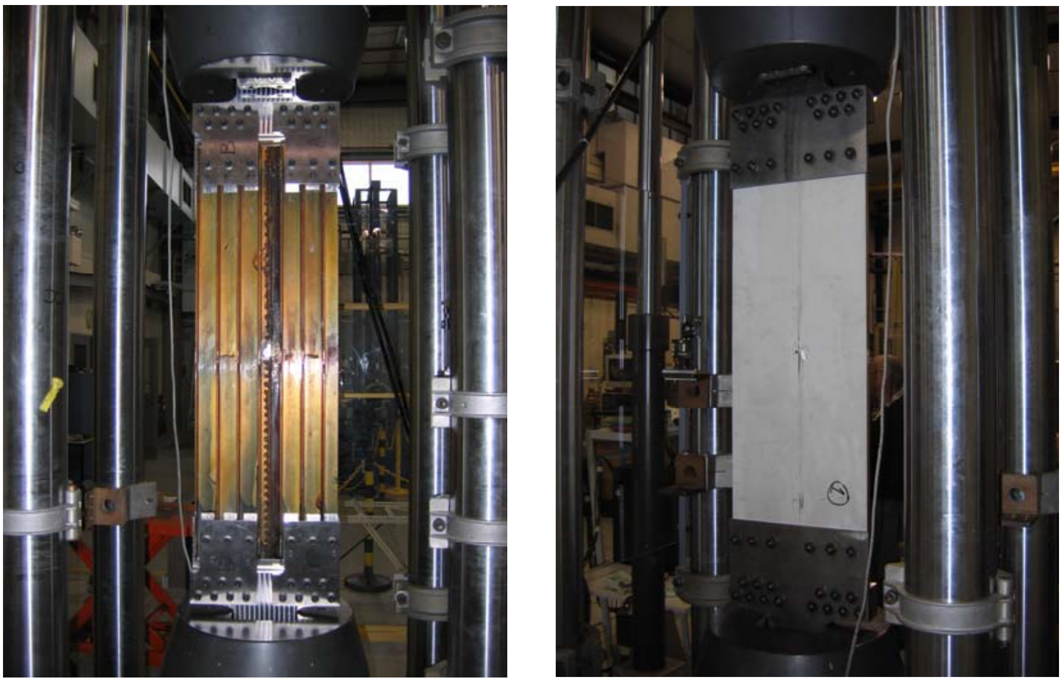

Figure 14 Wing splice panel component test set-up

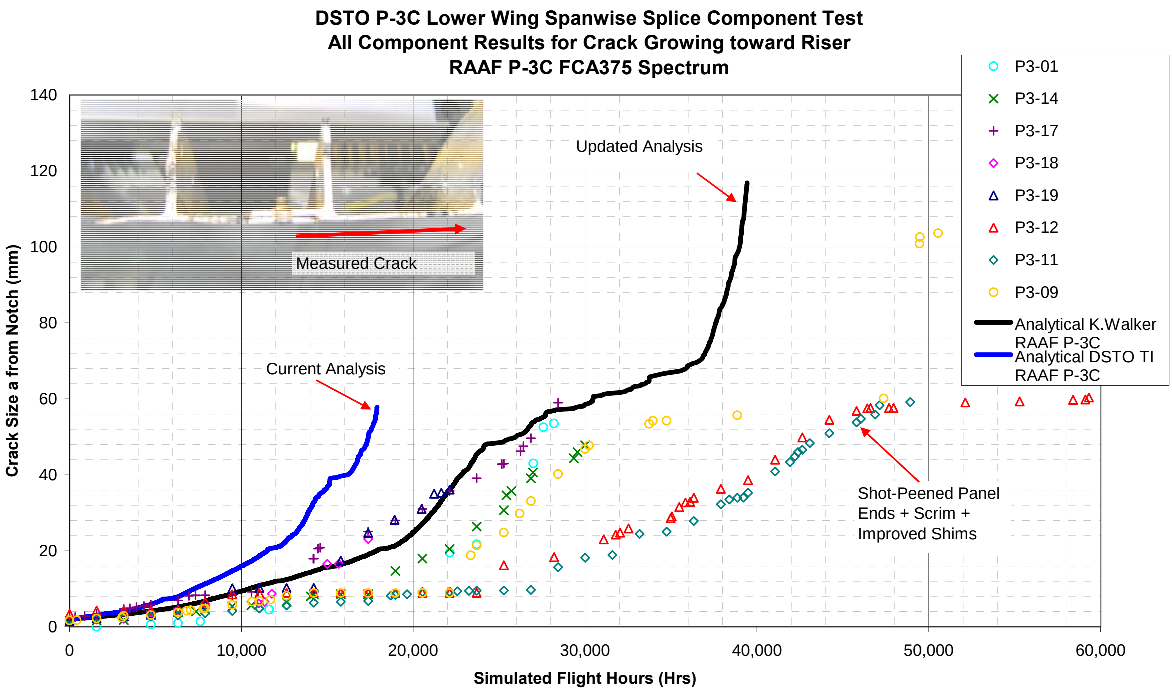

Figure 15 Crack growth results so far for span-wise splice component panels compared with current and updated FASTRAN analyses

8.2.11 P-3C Repair Assessment Manual *(C Lyons and S Vaskrsic [QinetiQ AeroStructures] and SQNLDR A. Bowler [RAAF])*

The Royal Australian Air Force (RAAF) operates 19 P-3C aircraft. The fleet is approaching the safe-life as defined by the original Certification Structural Design Standard (CSDSTD), CAR4b. As a consequence, a change of the CSDSTD for fatigue management to Federal Airworthiness Regulations (FAR) 25.571 has been approved.

Interpretation of FAR 25.571 requirements, supported by the guidance provided in associated Advisory Circulars (ACs) including AC120-93, has led to the decision to develop a Repair Assessment Manual (RAM). The RAM will provide procedures for engineers to assess both new and existing repairs applied to AP-3C aircraft, without the need for individual Damage Tolerance Analyses (DTAs). The output of the RAM will be inspection requirements for the repair. These include inspection thresholds and inspection intervals.

Initially, procedures to assess the repair condition will be provided. The repair will then be mapped to an appropriate Fatigue Critical Area (FCA). Many FCAs were identified following crack findings resulting from full-scale fatigue testing undertaken in the international collaborative P-3 Structural Life Assessment Program (SLAP). Each FCA has an associated inspection threshold and inspection interval.

Depending upon the criticality of the repair, the FCA inspection threshold and interval can be adjusted to provide inspection options for the repair. The crack initiation life is adjusted based upon a notch factor, K_N, comparison between the FCA and the repair configuration. The crack growth life is adjusted by assuming a cubic relationship between the crack growth rate parameter and far field stress.

The RAM is in the final stages of development. Validation of the procedures in the RAM, such as the cubic rule adjustment of crack growth life, is in progress. Also, there remain some areas where full compliance of the RAM with the requirements of FAR 25.571 is yet to be demonstrated. These issues remain to be resolved before RAAF acceptance of the RAM for fleet application. It is intended that the RAM will be implemented by the Through Life Support (TLS) contractor, Australian Aerospace (AA), to enable rapid repair assessment.

8.2.12 Support to RAAF C-130J Structural Integrity Management *(R Ogden and L Meadows, [DSTO]).*

In 1999 the RAAF enhanced its tactical airlift capability with introduction to service of the C-130J-30 Hercules. The certification basis for the 'J' model was predicated on the fact that it was not fundamentally different to earlier models. However the C-130J has more powerful, yet lighter engines and with some questions surrounding previous fatigue test outcomes, the RAAF were not convinced that such assumptions were valid as the basis for C-130J-30 certification. In particular non-compliances involved substantiation of the structural life of type and damage tolerance analysis and fidelity of the associated operational loads monitoring and individual aircraft tracking systems.

From an Aircraft Structural Integrity (ASI) management perspective the Commonwealth of Australia is currently focused on resolution of the outstanding structural certification issues, and establishment of mature data and system infrastructure to support ongoing management of structural integrity. Key activities aimed at addressing these issues are as follows:

<u>Collaborative Wing Fatigue Test</u>

A key activity aimed at substantiating the C-130J life of Type (LOT) involves a collaborative program with the UK Ministry of Defence to conduct a C-130J Wing Fatigue Test (WFT). Marshall Aerospace is the prime contractor for this activity with DSTO providing technical assurance for Australia as well as contributing directly to the program. The last 12 months in particular has seen a significant amount of work completed both in Australia and the United Kingdom in preparation for the start of test cycling in early 2009. In particular this involved the definition and development of a representative loads sequence (and subsequent conversion to actuator loads). More recently extensive work has been conducted to verify the representative response of the wing test article and of the simplified test article restraint, involving fuselage sidewall structure.

<u>Structural Health Monitoring & Operational Loads Monitoring</u>

In early 2008 the RAAF operational loads monitoring aircraft was released back into service and to date has collected in excess of 150 flights. Whilst satisfying RAAF airworthiness requirements to provide OLM fleet coverage the primary aims at this stage are to provide data for development of a RAAF representative loads sequence and to utilize OLM data to validate assumptions inherent in the on-board Structural Health Monitoring System (SHMS). It is anticipated that both full scale fatigue test and OLM data will be utilized to calibrate the existing SHMS.

<u>Full Scale Fatigue Test Interpretation</u>

A primary focus for DSTO over the last two years has been preparation for full scale fatigue testing and development of tools and processes to interpret the results. Interpretation of WFT damage findings and "read-across" to equivalent damage based upon an actual RAAF average usage spectrum is a fundamental component of the overall program. A key output, in compliance with the adopted certification basis, is the information for each critical location required to support structural integrity management of the C-130J-30, (i.e. principally the Safety by Inspection (SBI) program). This ultimately breaks down to the requirement for calculation of threshold inspection and recurring inspection information for each of the locations of interest. Contributing to this process are inputs from many other elements; spectra development, test data, test assumptions, material data, beta factors, damage model calibration, coupon testing (see *Figure 16*) and certification basis.

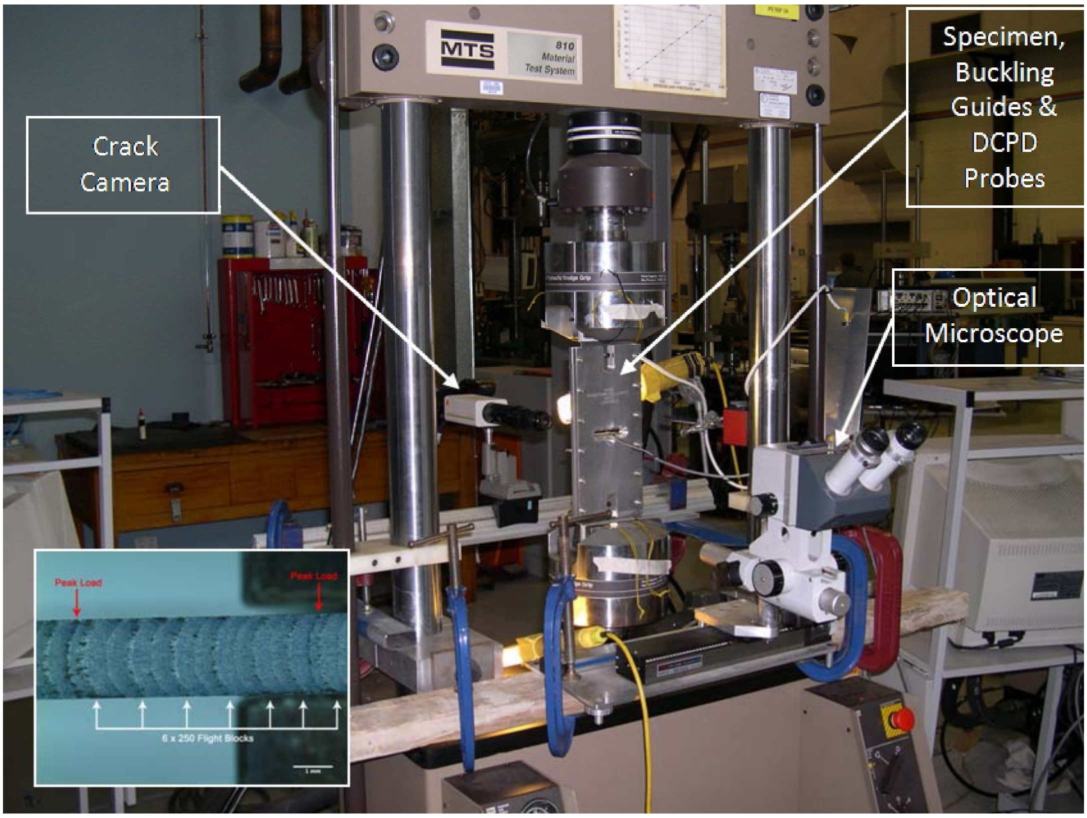

Figure 16 DSTO coupon testing and fractographic analysis (inset) in support of C-130J full scale fatigue test.

Interpretation

<u>Beta Solution Development</u>

A key component of the test interpretation work has involved the development of 3D stress intensity factor solutions, at key damage tolerance analysis locations on the airframe. These are obtained around the crack boundaries using advanced computational fracture mechanics codes (ESRD StressCheck). Some key features of this approach include: (i) adaptive mesh refinement via variable-order polynomial elements to achieve solution convergence, (ii) iterative non-linear modelling of contact, and (iii) parametric geometry definitions including crack shape. The parametric nature of the modelling allows multiple crack configurations to be considered at one location, to achieve solutions significantly more quickly than prior modelling approaches, (see *Figure 17*).

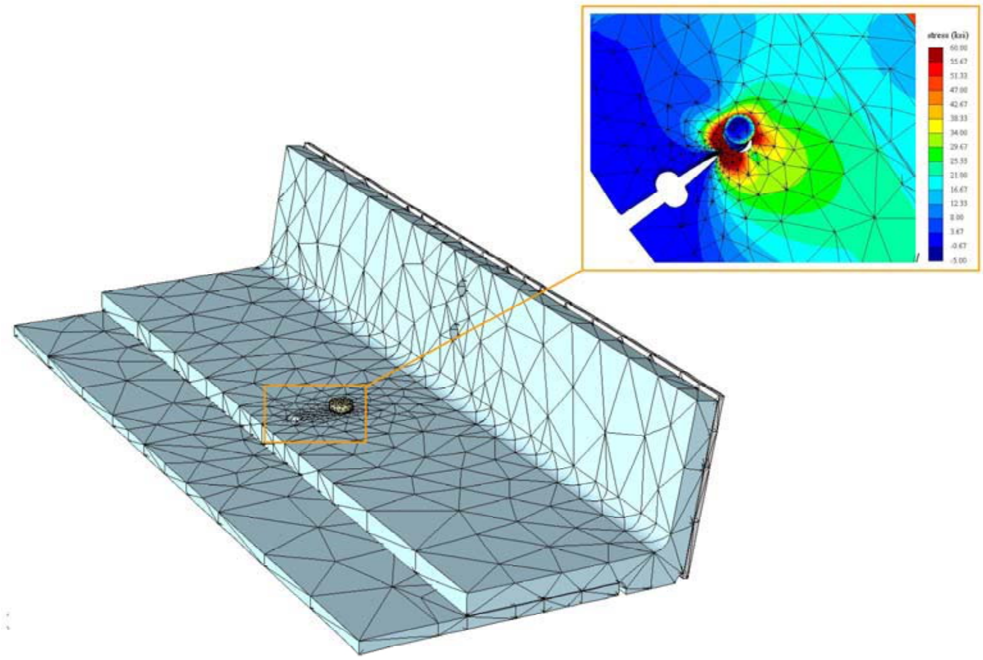

Figure 17 Lower Spar Cap FEM

8.2.13 RAAF PC-9/A Structural Life Assessment Study *(R Millard, S Wahono and K Watters [QinetiQ AeroStructures])*

The Royal Australian Air Force (RAAF) operates 64 Pilatus PC-9/A aircraft, which are used in advanced training, forward air control and aerobatic display roles. The aircraft entered service between 1987 and 1992, and as such are subject to various ageing aircraft issues typical of aircraft of this age and type. The PC-9/A Structural Life Assessment Study (SLAS) was initiated by Aircraft Structural Integrity-Directorate General Technical Airworthiness (ASI-DGTA) to assess the structure related safety, availability and cost of ownership aspects of PC-9/A operations. The study was conducted on behalf of ASI-DGTA by QinetiQ AeroStructures and was delivered in January 2008.

The key limiters of structural life of an aircraft (safety, availability and cost of ownership) were examined individually and in detail in the SLAS. The robustness of the fatigue certification basis was reviewed and any gaps in the fatigue management strategy were identified. Ageing aircraft effects including multi-site damage, widespread fatigue damage, repair interactions and corrosion compounding fatigue were also examined.

Historical condition records were audited to identify structural locations with frequent in-service damage, the defect types and historical damage trends. Analysis of this data was used as the basis to project future condition of the fleet. Condition projections, coupled with a review of the PC-9/A maintenance program allowed projections of future maintenance requirements to be made. The future availability of the fleet and its ability to achieve the required rate of effort were estimated from the maintenance projections. Different scenarios for corrosion prevention and control and Planned Withdrawal Date (PWD) were also considered.

The SLAS findings were drawn together to provide recommendations for addressing several minor certification basis deficiencies identified and for improving fleet availability and cost of ownership. The SLAS partially fulfilled the intent of an ageing aircraft audit, with its review of the certification basis robustness and fatigue management. It is also provided planning information to Training Aircraft Systems Program Office (TASPO) and RAAF capability management through its assessment of availability and cost, considering extended PWD scenarios.

8.2.14 Lead-In Fighter (LIF) Fatigue Test, *(I Anderson, [DSTO])*

In 1997, the RAAF formalised a contract for the delivery of 33 Hawk Mk.127 aircraft, to satisfy the requirement for a new Lead-In Fighter (LIF) aircraft. Under the terms of the contract, 12 aircraft were produced at Brough in the United Kingdom, with the other 21 manufactured in Brough and assembled at a purpose-built BAE SYSTEMS facility at Williamtown, New South Wales. The first UK-built Hawk Mk.127 made its maiden flight in December 1999, with the first Australian assembled aircraft following just five months later.

In comparison with the T Mk 1, the Hawk Mk.127 is approximately: 600 mm longer and 25% heavier (1% weight increase could equates to approx 5% reduction in life for unchanged structure), but is also significantly improved in capability in all key locations over the T Mk 1. DSTO had been asked to conduct an assessment to determine whether a full-scale fatigue test of the aircraft was required. Fatigue tests on earlier models of the Hawk had been conducted but the DSTO analysis determined that new tests were essential to provide specific information on RAAF operational usage of the Australian version of the aircraft, confirming the design adequacy and also minimising the possibility of fatigue-related accidents, and potentially increasing the operational life of the RAAF fleet.

Once the decision had been made to proceed with a comprehensive full-scale fatigue test, DSTO and BAE SYSTEMS entered a commercial agreement to conduct the test in Australia. Personnel from both organisations are involved in the test, as are several Australian contractors with a range of backgrounds and expertise. BAE SYSTEMS Brough Structural Test personnel designed the rig in conjunction with the BAE SYSTEMS Structures personnel. BAE SYSTEMS Structures personnel are responsible for the derivation of the FSFT Test Spectra and interpretation of any test results onto the Operational RAAF fleet, via consultation with the RAAF ASI-DGTA branch. DSTO are responsible for the assembly of the rig, the commissioning of the rig and specimen, the running of the rig in accordance with agreed timescales and quality requirements, inspections, damage reporting, and overseeing any repair activity. BAE SYSTEMS also use the FSFT results to support their worldwide Hawk fleet as appropriate.

The specimen comprises a LIF Hawk production standard Fuselage, Wing and Fin. A production standard Windscreen and Canopy are also fitted to enable cockpit pressurisation to be applied. Some structural items have been, or are being, cleared by an alternative route, and therefore are not fitted to the test specimen. Other items are present in dummy form, which means that the item is not being tested but that it is used to introduce load into the test specimen. Any items which are regarded as non-structural such as pipework, wiring, avionics boxes, etc, are omitted.

The loads spectrum for the Hawk LIF test is based on a mix of computational modeling and actual flight trials data. The most suitable mix was determined by BAE SYSTEMS, using the extensive Hawk data already available as well as the RAAF Mk 127 specific flying data accumulated for this purpose, thereby combining all appropriate data to create a robust FSFT spectra covering both the contract requirements and the future anticipated RAAF Operational flying.

The test at Fishermans Bend is being complemented by a rear fuselage and tailplane test applying both flight manoeuvre and buffet loads. The tailplane test is being conducted in the UK by BAE SYSTEMS.

The LIF FSFT test rig, *Figure 18*, is located in a new test laboratory at DSTO's Fishermans Bend site, it is an 8-metre high, three-level test rig designed by BAE SYSTEMS with DSTO input. Key features include:

- a 1st level platform raised 2.0 metres off the ground to allow sufficient headroom and systems installation within the rig footprint;
- additional floor area on all levels surrounding the fatigue article to allow ergonomic work practices and equipment distribution;
- a removable top deck that allows for the installation and removal of the fatigue article as required.

The main control system was developed by MTS Systems Corporation, USA. DSTO provided background intellectual property to enhance the functionality of the system. The system applies and monitors loads to the test article from 84 hydraulic and 6 pneumatic channels simultaneously. It also includes a fully integrated 1200 channel data acquisition system. The hydraulic actuators were designed and manufactured by MOOG Australia, while the valve packs, or Controlled Abort Manifolds, which form part of a fully independent controlled abort unload system, were also produced by MOOG Australia, although based on DSTO design principles. The associated unload controllers were developed jointly by Moog and MTS.

Design, construction and commissioning of the test rig was completed in December of 2005. The fatigue test commenced in December 2005 with the application of a simulated Production Flight Acceptance Test (PFAT) load block. Continuous testing commenced in February of 2006 and is expected to continue until 2012, eight years before the fleet's proposed withdrawal date. The fatigue test will be followed by DSTO carrying out a residual strength test and a full teardown inspection. Overall the project will be completed by 2014.

Figure 18 LIF Test Rig (left) under construction, and (right) with aircraft installed.

The Hawk Mk.127 fleet has been operational since 2000, under an interim flight clearance that allows for such activity prior to full-scale testing. Fatigue testing commenced in February of 2006. The first objective is for the test to clear the fleet aircraft for 3000 flying hours, as defined by the contract spectrum. Because the FSFT spectrum is a robust one covering future RAAF Operational activities as well as the contract requirements, test "read across" carried out by BAE SYSTEMS shows differing clearances for different parts of the structure when compared with how the RAAF fleet are currently flying. The 3000 flying hours clearance required by the RAAF has already been achieved by the FSFT, and indeed it is currently at 12,150 test hours of a total test duration requirement of 50000 test hours. The 50000 test hours requirement will give adequate clearances to all structure, whether monitored or unmonitored, of the contractually required spectrum.

The test article is currently undergoing a major inspection / rework program, which involves removal of the wing to incorporate several modifications that are also to be implemented in the RAAF Hawk LIF fleet. The test is scheduled to re-start in May 2009.

8.2.15 Risk Analysis Methodologies *(K Watters, P Livingstone [QinetiQ AeroStructures] and R Antoniou [DSTO])*

QinetiQ AeroStructures has been tasked by DSTO to develop methodologies and capabilities in risk analysis of aircraft structural fatigue failure. The work included a comparative review and assessment of the risk basis of the principal airworthiness standards used by the RAAF for airframes and engines (DEFSTAN 00-970 (Parts 1 and 11), FAR 25, FAR 33, JSSG-2006 and JSSG-2007). The work also included developing a methodology to analyse whole system risk of failure, with a focus on engine structures. A significant element of that work was an assessment of the impact of batch-to-batch variation of aircraft materials or production processes on fleet risk of failure.

Another part of the work has been to develop a methodology and capability to determine the risk of failure of an aircraft that has been initially certified and managed under a safe-life philosophy but has been transitioned wholly or partially to Safety-By-Inspection (SBI) management. A number of RAAF aircraft types (e.g. P-3C Orion) fall into this category. It allows the aircraft to be operated beyond its safe life to achieve a planned withdrawal date.

The methodology for analysing transition to SBI largely follows standard methods for modeling the variability in the growth of fatigue cracks from initial flaws and allowing for the probability of their detection during inspection. The

significant development of the work has been to adjust the means of the probability distributions for initial flaw size and crack growth rate to match full-scale test data, and to adjust the standard deviations to match a standard value, such as 0.1296 from DEFSTAN 00-970 for highly loaded parts of airframe structures.

Through this approach, the risk analysis matches a safe-life analysis during the safe-life management phase and then extends that into the SBI phase. This overcomes a principal concern with risk analyses, which is whether the selected probability distributions for initial flaw size and crack growth rate include the full range of flaw behaviour under in-service environmental conditions. It is expected that it should also render the risk analysis less sensitive to choice of probability distribution type, although that remains to be investigated. The methodology has been demonstrated for a simple case study, and the next stage would be to analyse a RAAF fleet example.

8.2.16 Failure Analysis Examples – Military Aircraft *(N Athiniotis, [DSTO])*

The following sections contain examples of failure investigations conducted in the last two years by DSTO:

8.2.16.1 RAAF C130H nose landing gear strut failure

During a Touch-and-Go, the crew observed a bang and nose wheel shimmy when the nose wheel made contact with the ground. They continued with the Touch-and-Go, and once airborne the NLG was inspected by the crew through the inspection window, with visible cracking observed. A safe landing was subsequently conducted.

The DSTO investigation found that the failure was caused by the presence of a large pre-existing defect that may well have been present since manufacture, see *Figure 19* RAAF C130H nose landing gear strut failure The large defect was most likely a quench crack, formed during heat treatment. Evidence revealed chrome near to the surface of the steel, indicating that the defect was present the last time the component had been stripped and replated with chrome.

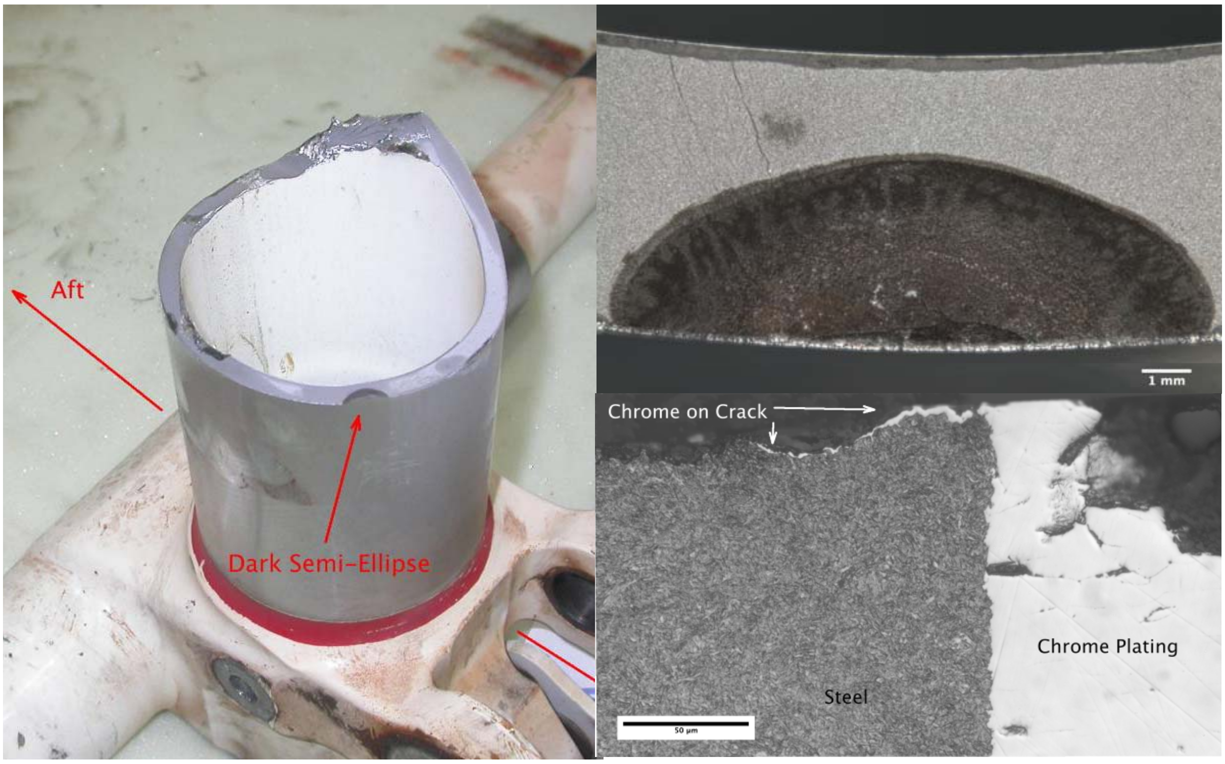

Figure 19 RAAF C130H nose landing gear strut failure.

8.2.16.2 RAAF PC9/A 1st stage reduction gearbox sun gear failure

Routine oil filter analysis revealed the presence of large amounts of alloy steel debris. Subsequent teardown and inspection of the engine revealed that the 1st stage sun gear had failed due to spalling, Figure *20*.

Microstructural analysis indicated that the most likely cause of the spalling was rolling-contact fatigue due to excessive loads on the working surfaces of the splines. While the components had been manufactured using the correct materials, the hardness test results indicated that the case hardening of the sun gear was borderline, and the effective case depth was below the specified minimum. This may have been a contributing factor to the onset of failure.

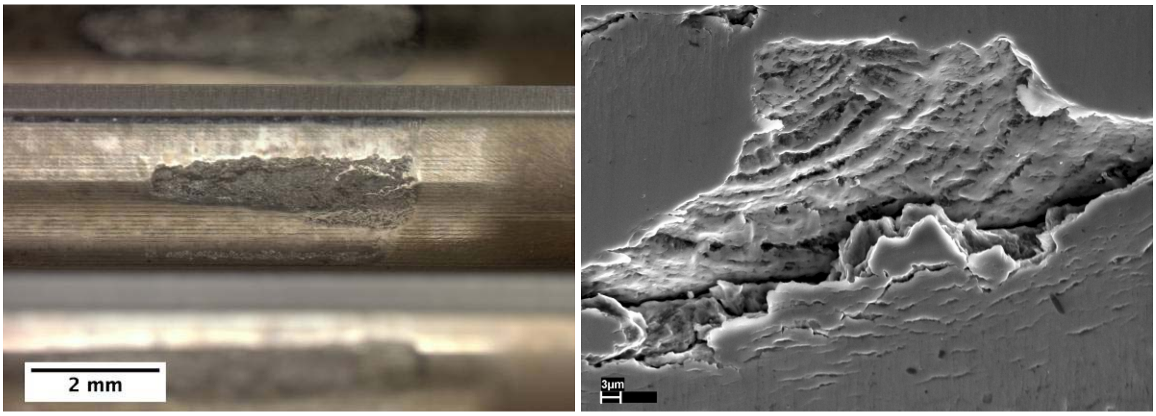

Figure 20 RAAF PC9/A 1st stage reduction gearbox sun gear failure.

8.2.16.3 RAAF F/A-18 Nose landing gear up-lock mechanism failure

Fractographic examination indicated that fracture of the F/A-18 NLG up–lock mechanism was caused by the initiation and propagation of multiple fatigue cracks. Final tensile fracture of the up–lock mechanism ensued when the fatigue crack reached critical size and the uncracked ligament was no longer able to carry the applied loads.

Further optical fractographic examination revealed that the fatigue cracks initiated from surface pits. The appearance of the pits resembled those formed during chemical etching of aluminium surfaces, *Figure 21*.

The fatigue cracks propagated to approximately 10 percent of the total fracture surface indicating that the component was subjected to very high nominal applied stresses.

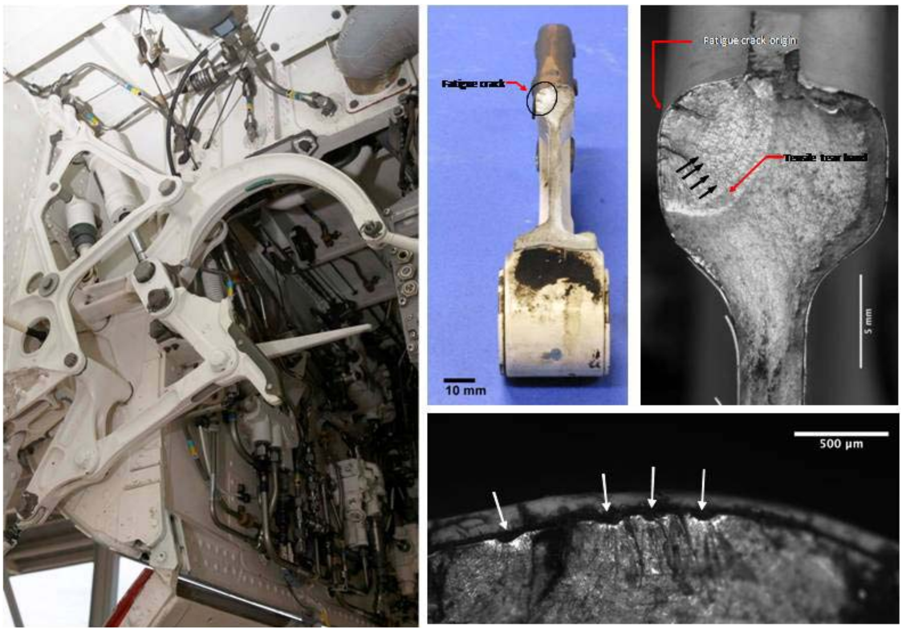

Figure 21 RAAF F/A-18 nose landing gear up-lock mechanism failure

8.2.16.4 Black hawk high speed shaft tube cracking

The investigation revealed that excessive contact between the fasteners used to hold the tube in position and the bore of the fastener holes of the support flange would have resulted in point loading and flexing of the support flange. Furthermore, the evidence suggested that the fasteners had been forced into position and possibly used to drive the Front Support tube itself into its installed location.

It was likely that the combined effects of the service stresses, the point loading (and flexing) and the presence of surface damage and corrosion pits caused the fatigue crack initiation, Figure *22.*

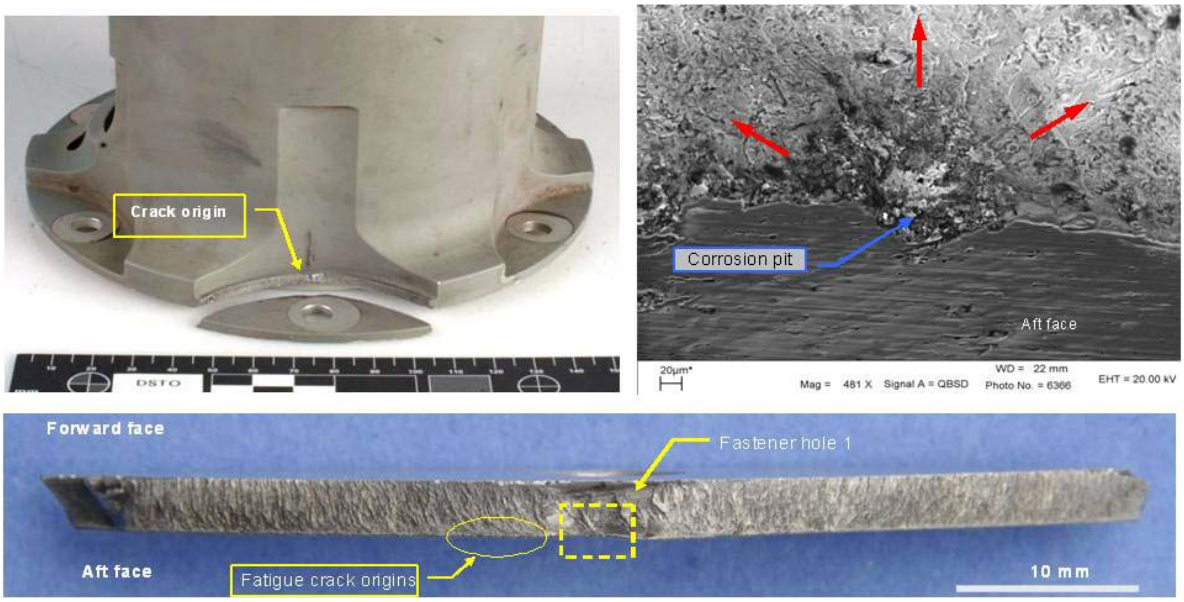

Figure 22 Black hawk high speed shaft tube cracking.

8.3 FATIGUE OF CIVIL AIRCRAFT

8.3.1 First Diamond: 'Damage tolerance' for the structural honeymoon. *(S Swift [Civil Aviation Safety Authority])*

Paper to be presented at ICAF 2009 Symposium

Abstract

This paper is the third in the 'diamond' series on *damage tolerance*. The first was *Rough Diamond*, in 2005. It introduced the 'diamond', a simple division of damage tolerance into five essential 'facets': *site*, *scenario*, *detectable*, *dangerous* and *duration*. The second was *Rusty Diamond*, in 2007. It argued the diamond's applicability to corrosion (or any progressive damage), not just fatigue.

Now, this is *First Diamond*. In the same tight, clear style, it tackles the thorny issue of the *threshold*: the end of the structural 'honeymoon', the beginning of the inspection program. Thorny? Yes, the threshold is easily the most contentious aspect of damage tolerance. It is even contentious for the two biggest airworthiness authorities and the two biggest aircraft manufacturers. Airbus' criticism of FAA policy at ICAF 2005 is just one example.

Contention is not surprising. First, it is the money at stake. Late thresholds save maintenance for airlines and boost sales for manufacturers. Second, is the sensitivity to manufacturing quality. Flaws only have to increase by millimetres to *decrease* thresholds by thousands of hours. Third, is the obsession with *prescription*. Prescribing, exactly, manufacturing quality (type and size of flaw) and analysis method (like fracture mechanics) frustrates those who could get the same safety *outcome*, with a better threshold, in other ways.

On 'safety', if we prescribe manufacturing quality, as just described, what is the incentive to improve it? Or, to consider other flaws or damage, possibly more likely or more dangerous?

First Diamond offers ideas to improve thresholds:

- Damage tolerance
  - use the 'diamond' *exactly as for intervals*
  - to improve simplicity and consistency
- Risk management
  - consider all risks (not just fatigue), prioritised by chance and consequence
  - to improve safety and efficiency
- Human factors
  - to temper conservatism, which is counter-productive if excessive
  - to improve credibility and enforceability
- Outcome-based rules
  - brief, broad safety outcomes
  - to improve clarity and flexibility

The author argues and illustrates his ideas with examples drawn from 25 years regulating structural maintenance programs (including thresholds) for one of the world's oldest and hardest-working fleets. He also reviews the work of others, especially the USAF and the promoters of HOLSIP.
He then considers special cases:

- *Escalated* thresholds
  - Are 'nil findings' relevant?
- *Unnecessary* thresholds
  - Could a threshold cost more than it saves?
- *Unpublished* thresholds
  - Is it safe to be secretive about post-DSG (Design Service Goal) thresholds?
- *Enhanced* thresholds
  - When are cold working and other fatigue counter-measures creditable?
- *Combined* thresholds
  - Should thresholds for fatigue and corrosion be separate or combined?

• *Composites* thresholds
 − How are they different?
• *Healthy* thresholds
 − How will Structural Health Monitoring affect thresholds?
• *Life limits* and thresholds
 − How are they alike and what are the implications?

The paper hopes to promote safety, economy and harmony in our use of thresholds.

8.3.2 Bridging the gap between theory and operational practice *(S Swift [Civil Aviation Safety Authority])*

'Bridging the gap between theory and operational practice' is the theme of ICAF 2009. So, Steve Swift from Australia's Civil Aviation Safety Authority (CASA) asks how well we are bridging the gap for these four theories:

- The theory of the 'limit of validity'
- The theory of fatigue analysis
- The theory that corrosion inhibitors affect fatigue
- The theory of the 'rogue flaw'

<u>*The theory of the limit of validity*</u>
How well are we bridging the gap for the theory of the 'limit of validity' (LOV)?

In 2006, in the US *Federal Register* (Volume 71, No. 74, page 19930), the FAA described the theory:

> *The ARAC (Aviation Rulemaking Advisory Committee) recognized that structural fatigue characteristics of airplanes are only understood up to a point in time consistent with the analyses performed and the amount of testing accomplished. The maintenance program inspections related to structural fatigue are based on the results of these analyses and tests. Therefore, these inspections may need to be supplemented by further inspections, modifications, or replacements, if operation beyond a certain point is planned. The ARAC recommended that there should be a 'limit of validity of the maintenance program' to limit the operation of an airplane. Once an airplane reached this limit, the operator should no longer operate the airplane, unless the operator has incorporated an extended limit of validity and any necessary service information into its maintenance program.*

The theory is not yet US law. But, already, one US aircraft manufacturer, Cessna, voluntarily includes a LOV in each of its structural maintenance programs. They challenge if regulators want to enforce the theory or only write about it. In 2007, Cessna published a LOV for their 441 twin-turboprop. It was less than the age of several Australian-registered 441s. Some were in airline service.

CASA felt it had no choice but to ground them.

Figure 23 Cessna 441s that CASA grounded because they exceeded the LOV for their maintenance program

The FAA is still deliberating. It has more time. Its Cessna 441s are only half the age.

The theory of fatigue analysis
At ICAF, we often discuss the theory of fatigue analysis. We debate the relative merits of cumulative damage and fracture mechanics. It might be sobering to consider a recent judgement, by the world's largest civil aviation safety regulator, on *both* theories of fatigue analysis.

FAA Advisory Circular 91-82, *Fatigue Management Programs for Airplanes with Demonstrated Risk of Catastrophic Failure Due to Fatigue*, says, on page 3:

An airplane type design has a "demonstrated risk of catastrophic failure due to fatigue" when:

- *An airplane has experienced a catastrophic failure due to fatigue and the same scenario is likely to occur on other airplanes in the fleet,*
- *Airplanes of the type design have a service history that indicates a significant likelihood of catastrophic failure due to fatigue in the fleet, or*
- *Fatigue testing of the type design indicates a significant likelihood of catastrophic failure due to fatigue in the fleet.*

Analysis is not on the list. The FAA does not yet trust either cumulative damage or fracture mechanics, at least not enough to put a fatigue management program they predict is necessary for safety into operational practice with an Airworthiness Directive.

Does it matter? Do we not always verify analysis by test? Often, for small (FAR 23) aircraft, we do not. Often, there is only analysis. So, often, for small aircraft in the USA (unlike in Australia), there is no Airworthiness Directive for fatigue management. Let us hope the analysis *is* wrong.

Figure 24 The Piper Chieftain has an Airworthiness Directive for fatigue management in Australia, but not in the USA

The theory that corrosion inhibitors affect fatigue
CASA is sponsoring research into the use of corrosion inhibiters for small (FAR 23) aircraft, including small airliners, where their use is less common and less well understood than for large (FAR 25) airliners. The research, by RMIT and Monash Universities, will test the theory that corrosion inhibitors worsen fatigue in joints. The theory says they could lubricate and lessen load transfer by friction. Is it more or less of a problem, in operational practice, than the corrosion they prevent?

Past research has focussed on big, heavy joints for major airlines and the military. The aim is to explore the effect for the smaller and lighter world of general aviation, charter and regional airlines.

Figure 25 The Cessna 210 has a wing carry-through forging that is prone to corrosion as well as fatigue

The theory of the 'rogue flaw'
At ICAF 2009, in the third of his 'diamond' series of papers, one of the things Steve Swift looks at is bridging the gap between theory and operational practice for the 'rogue flaw'. Called *'First Diamond, damage tolerance for the structural honeymoon'*, the paper is about setting inspection thresholds. It discusses the theory that it is always conservative to assume a 'rogue flaw' and 'grow' it by fracture mechanics.

It concludes that, in operational practice, the rogue flaw is not always conservative or desirable. It can cause neglect of non-crack-like damage, which can be just as dangerous and likely. It can cause overconservatism, which can cause the threshold to lose so much credibility that operators ignore it. We should use it with care.

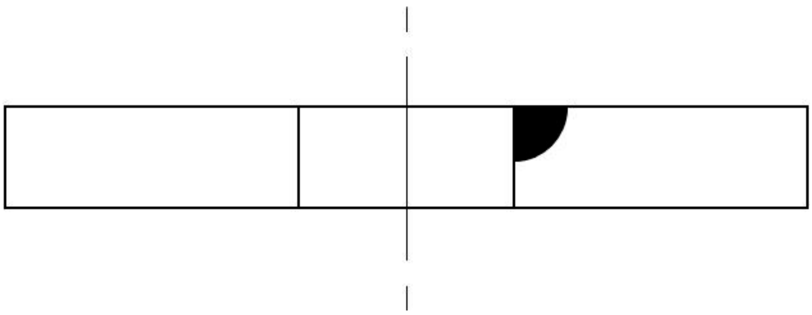

Figure 26 The rogue flaw – is it always conservative?

8.3.3 An evaluation of a full-scale fatigue test on a Nomad aircraft *(C Nicholson [Boeing Australia])*

A re-evaluation of fatigue tests on a complete Nomad airframe was completed and has been accepted by CASA and a service bulletin was issued for performing visual inspections without fastener removal. The re-evaluation had the aim of developing more suitable inspection techniques. The fatigue test had been completed during the 1980s and had produced apparently anomalous results in respect of one component, the Nomad stub wing (Figure 27), where three

components in total failed all at the identical location on the 2024 spar cap (Figure 28), two at approximately 130,000 hours, and one at 37,000 hours, compared to a projected life of 300,000 hours. The failures were significant since the basic aircraft configuration resulted in the stub wing being a primary load path for flight and ground loads. The configuration also resulted in the fatigue spectrum being dominated by a large Ground-Air-Ground (GAG) cycle which also featured large compressive stresses during the ground portion.

Firstly, through analysis of coupon data kindly supplied by FTI in Seattle, it was possible to show that the 130,000 hours was in fact a realistic estimate for the component as the coupon data showed that the compressive stresses in the GAG contributed significantly to the accumulated fatigue damage. Indeed for stress ratios of the order of R=-2, 7075T6 aluminium alloys possessed better fatigue properties than the 2024T3 series alloys.

It was also possible to develop a fracture mechanic approach which could match analysis to the fractographic records which had been produced as a result of the component failures at the time of the fatigue tests. These crack scenarios considered double crack growth from a rivet hole, which when it broke through to near edge transitioned to a single crack with a much larger characteristic crack length. It was also possible to postulate that the premature failure at 37,000 hours resulted from initial flaws in the holes of the test structure, and that during the first test an unrelated failure in the test rig produced a significant overload, which resulted in crack growth retardation.

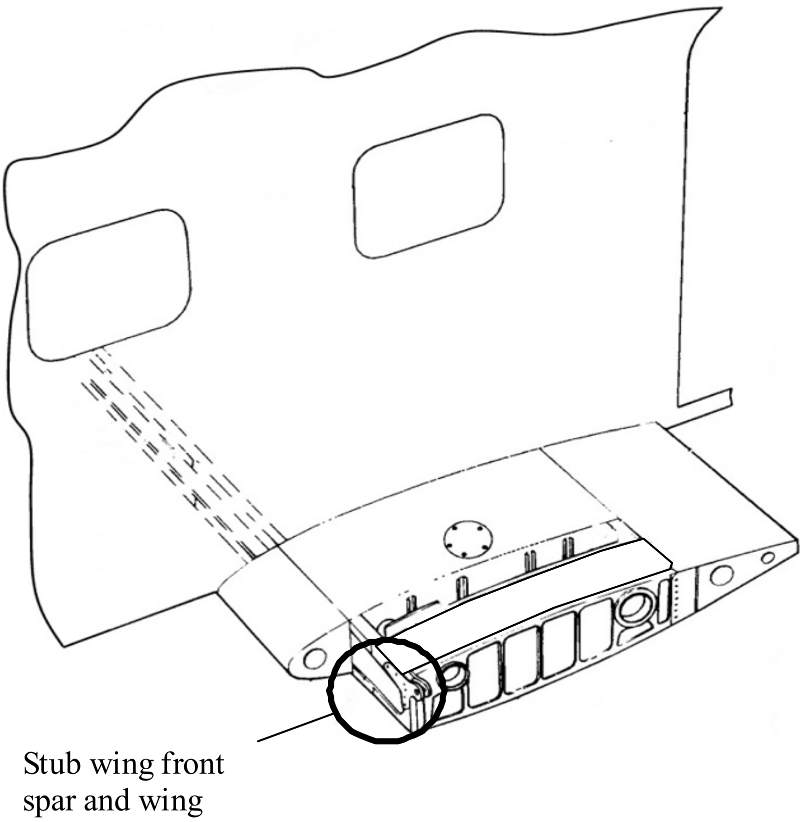

Figure 27 Nomad Stub Wing Structure.

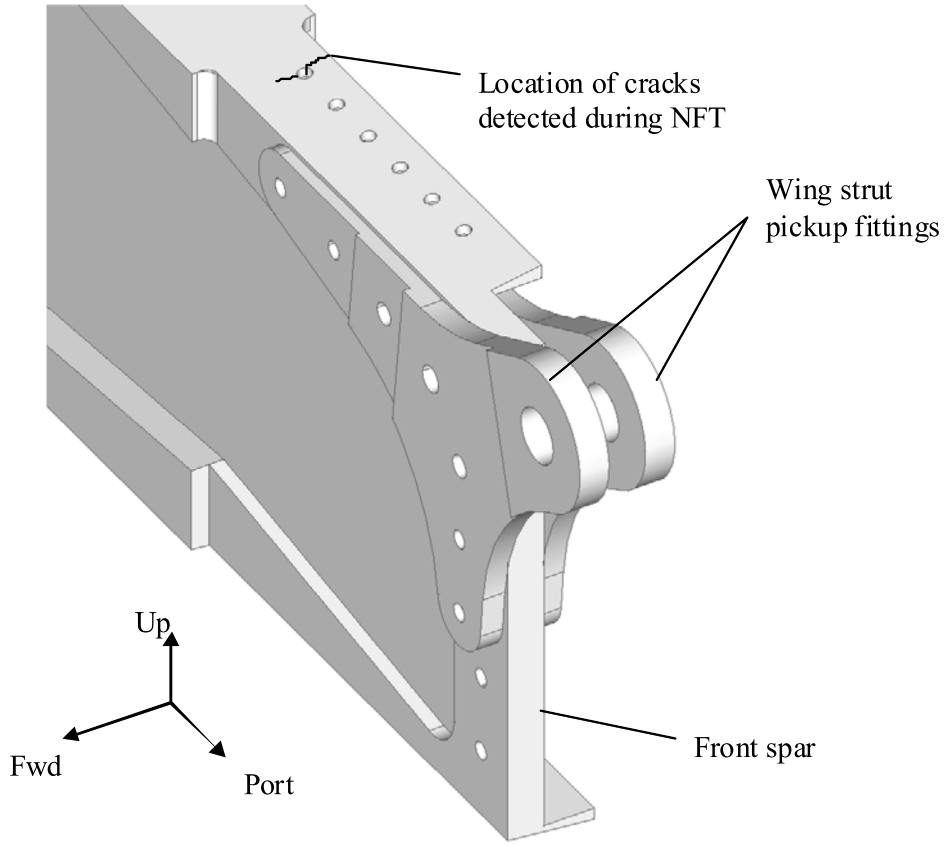

Figure 28   Front Spar.

8.4 FATIGUE-RELATED RESEARCH PROGRAMS

8.4.1 Loads-based fracture surface marking to aid quantitative fractography (QF) of fatigue in metallic materials *(S Barter, P White [DSTO], R Wanhill [NLR Netherlands])*

Paper to be presented at ICAF 2009 Symposium

Abstract

The selection of fracture surface marking methods based on exploiting or altering the required fatigue loads is of much interest for many fatigue test programmes. This is particularly true when crack growth measurements during testing are not possible or insufficiently accurate. In such cases, and as a check on crack growth measurements taken during testing, post-test Quantitative Fractography (QF) of fatigue crack growth is an invaluable tool. The QF can be made possible and/or greatly facilitated by fracture surface markings that can be related to loading events during the testing.

This paper reviews and discusses many examples of fatigue loadings that create fracture *surface* markings both naturally, as sometimes happens, and intentionally. These examples are from fatigue life tests of aircraft alloy components and specimens, particularly high strength aluminium alloys, under normal environmental conditions (air at ambient temperatures).

To better understand the visibility of markers, which is obviously important, a model is presented of how some aluminium alloy fatigue fracture surface features are produced by certain load combinations. This model is still being developed, and should lead to improvements in numerical modelling of the fatigue crack growth process.

The advantages and disadvantages of the intentional marking methods are also discussed with a view to obtaining guidelines and procedures for optimising the fatigue fracture surface topographies for QF.

8.4.2 Evaluation of the C Star Model for Addressing Short Fatigue Crack Growth *(K Walker and W Hu, [DSTO])*

The C* model has been proposed to account for the breakdown of K-similitude which occurs for short cracks. The model is based on the concept that crack growth rate is dependent not only on the stress intensity range, but also on crack length. The C* model was evaluated using experimental data from the open literature. For comparison, two other models, the El Haddad model and the FASTRAN model, were also evaluated for their capability in dealing with the same problem. The objective of the evaluation was to assess the performance of the C* model in comparison with the other two models in treating long and short crack growth in a unified manner, and to illustrate their merits and shortcomings. The developers of the C* model claim that it addresses the breakdown of similitude, which occurs typically in the threshold and near threshold region (Region 1) on a rate versus stress intensity plot. The phenomenon is clearly demonstrated in *Figure 29*, where the short cracks grow at applied stress intensity levels below the long crack threshold, and for a given applied stress intensity range they grow at a faster rate. There is also considerably more scatter in the data. A model which addresses this breakdown of similitude would be able to collapse these differences, at least to some extent. However, as shown in Figure 30, the C* model was not able to do so, with the long and short crack data (both from Region 1) still displaying a distinct difference from each other. This difference is readily evident, even over and above the scatter within the individual data sets. If the C* parameters determined by this process are used to predict the crack growth for a particular case, they produce very different predictions as shown in *Figure 31*. This is an undesirable characteristic, and does not support the proposition that the model addresses breakdown of similitude. The overall conclusion from the work was that, for the cases tested, the C* model was found to be ineffective in resolving the issue of breakdown of similitude for short cracks, and was of less practical use than the other models.

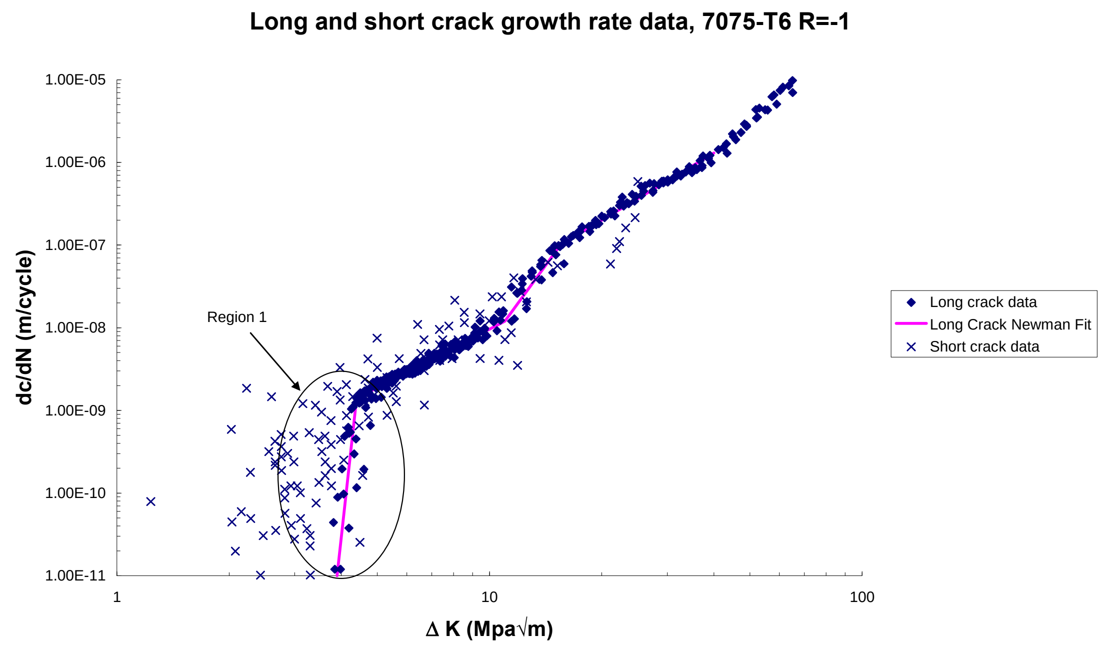

Figure 29 Long crack growth rate data for 7075-T6 material

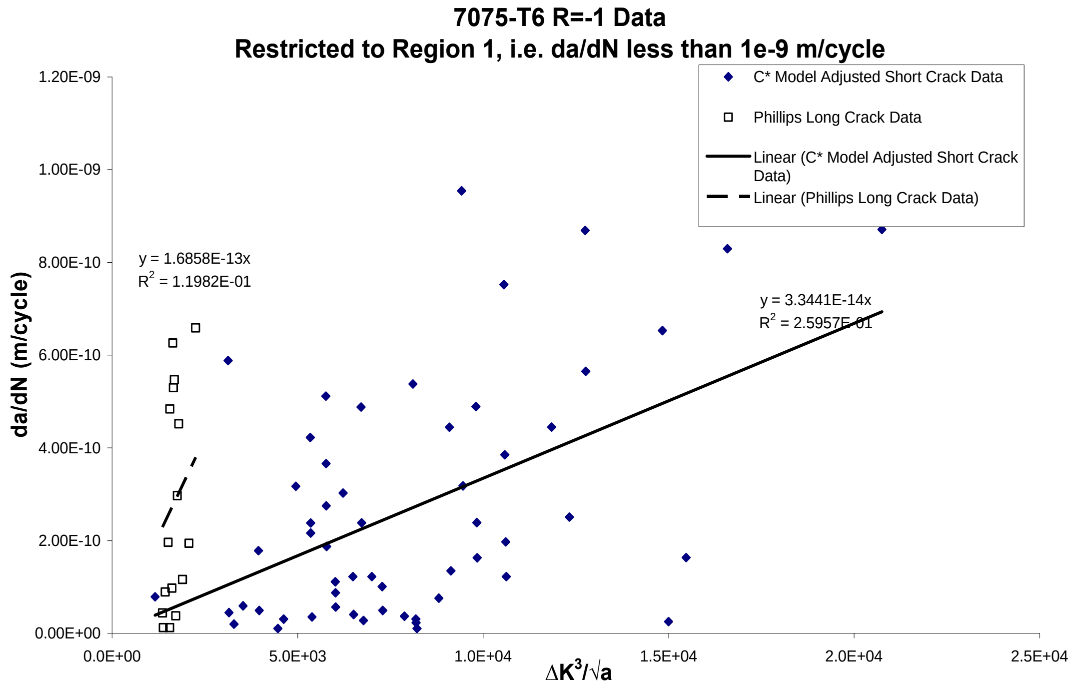

Figure 30 Crack growth rate plotted against the C parameter. Note the difference between the long and the short crack data points.*

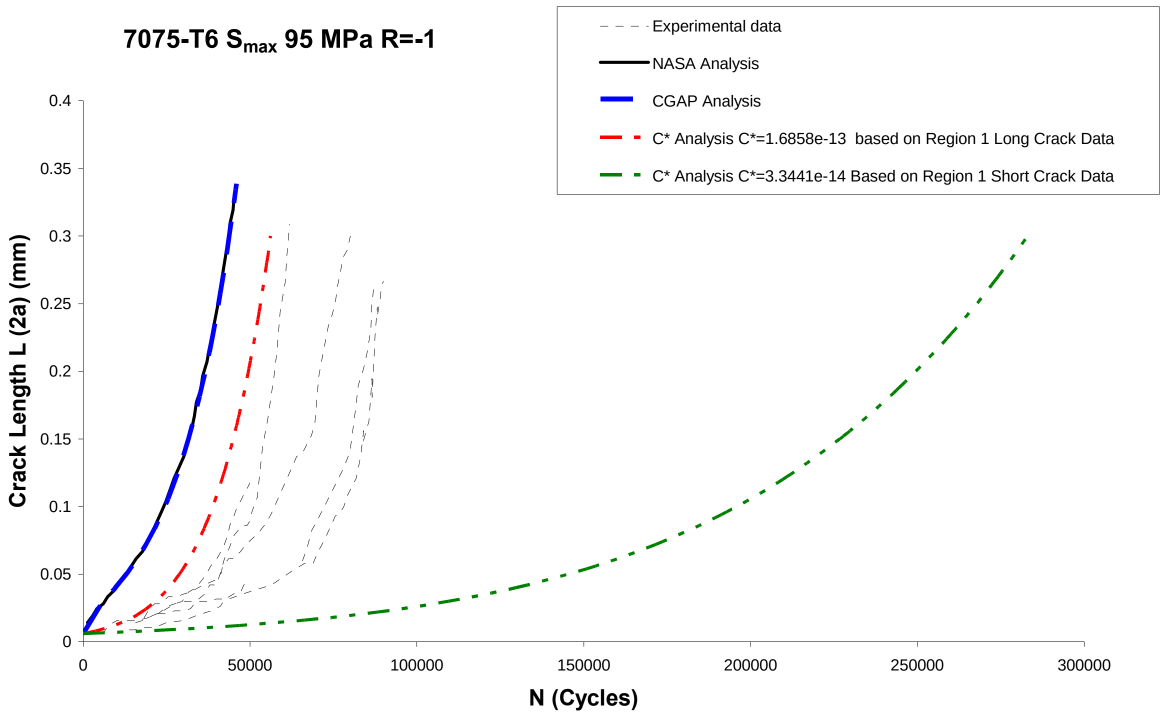

Figure 31 C based crack growth analysis results using long and short crack data restricted to Region I compared with the experimental crack growth data.*

Reference:

Walker, K., and Hu, W., "Evaluation of the C* Model for Addressing Short Fatigue Crack Growth", DSTO-TR-2185, October 2008

8.4.3 A study of the growth and coalescence of micro surface fatigue cracks in aluminium 7050. *(W Hu, Q Liu and S Barter [DSTO])*

Paper to be presented at ICAF 2009 Symposium

Abstract

Multi-site cracks may occur in structural components with shallow notches; in the early stages of fatigue damage, in corroded parts, or as a result of multiple notches such as in fuselage lap joints. In these cases, the development, interaction and coalescence of these multiple cracks poses a challenge to the accurate prediction of the rate of fatigue crack growth and the residual strength of these components, as the interaction of cracks tends to accelerate their growth, and the resulting combined crack usually leads to a lower residual strength earlier in life than cracks that grow individually. It is, therefore, important to gain an insight into the nature of interaction and coalescence of multiple small surface cracks and to investigate the criteria by which coalescence takes place.

In this paper we analyse the interaction and coalescence of multiple micro surface cracks in plates of 2024-T3 and 7050-T7451 aluminium alloys. Earlier studies show that for 7050-T7451, which is used in combat aircraft such as the F/A-18, the initiation and growth of micro-cracks can consume 50~90% of the total fatigue life of components, hence it is vital to accurately assess the evolution of these micro-cracks and their link-up to form primary long cracks. Experiments with the above materials have been carried out under constant amplitude cyclic loading and spectrum loading, where test coupons contained artificially generated micro-cracks (focused ion beam cut notches) with a surface length of ~50 µm and a depth of ~15 µm. These parallel shallow notches were arranged co-linearly or with an out-of-plane offset, with varying distances between their tips and offsets. The crack length and the time of coalescence of cracks growing from these notches were measured using quantitative fractography, as shown in the Figure 32 below.

Using linear elastic fracture mechanics, finite element analyses were performed to determine the extent of interaction between cracks and the condition under which two cracks may be considered to have coalesced. This was investigated by analytically varying the distance between the inner crack tips and the offset. These results were compared to the experimental results. Some analytical results are shown in Figure 33. Using these analytical results in a plasticity-induced crack closure model, modified to incorporate the use of the El Haddad model to account for plasticity when calculating the stress intensity factor solution, fatigue crack growth prediction was made for different service load spectra, and compared to the experimental results.

It is envisaged that the interaction and coalescence model developed in this work will help to improve fatigue life assessment in the short crack regime for the aluminium alloys considered.

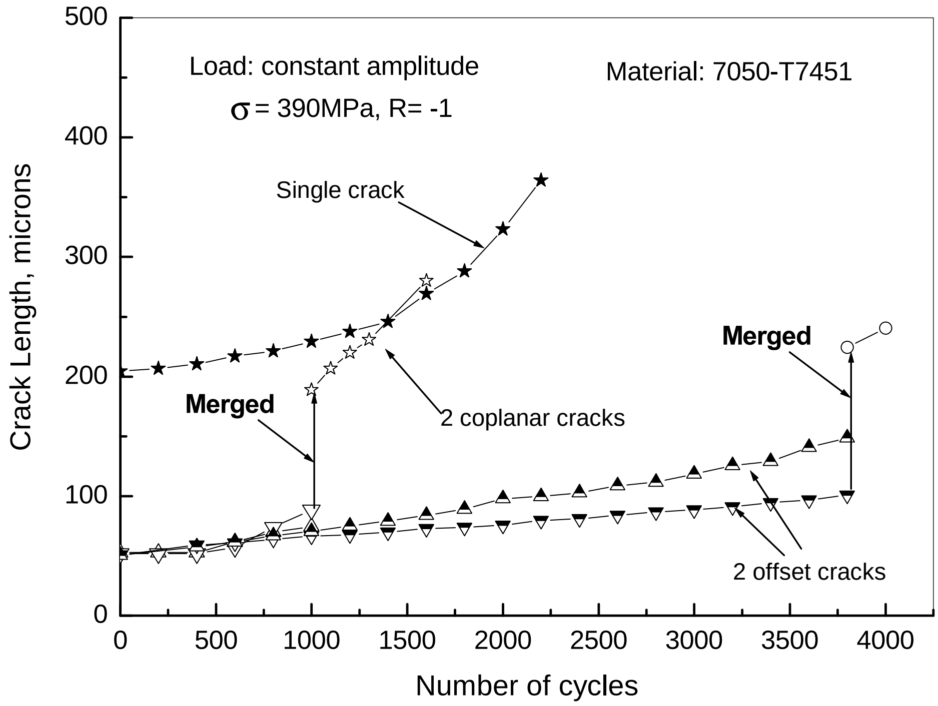

Figure 32 Experimental crack growth curves for coalesced and individual cracks

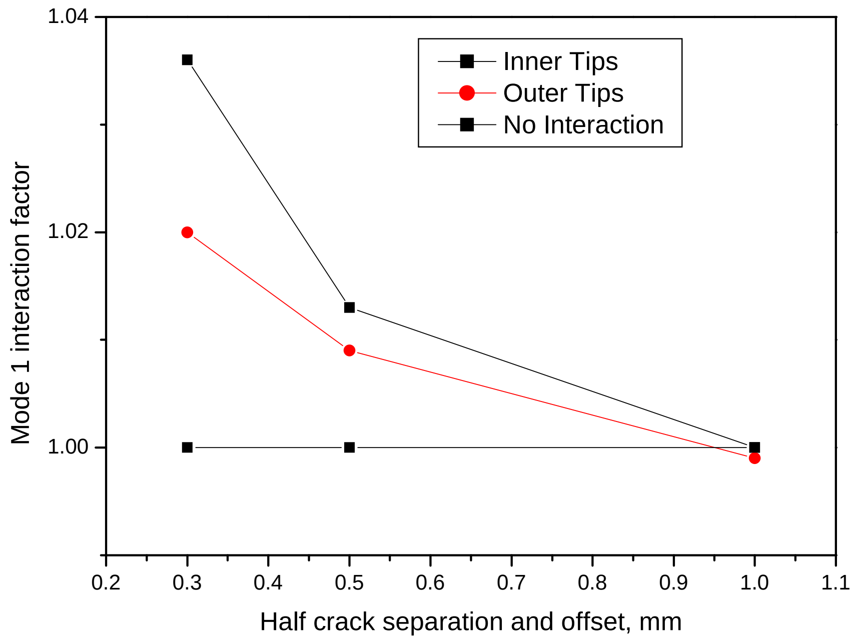

Figure 33 Numerical results showing interaction between parallel surface cracks with an offset. The interaction factor is defined as the ratio of the stress intensity factor for the multiple cracks to that of the single crack.

8.4.4 An evaluation of the effective block approach using P3-C and F-111 crack growth data *(C Wallbrink, R Amaratunga, W Hu, P Jackson and D Mongru [DSTO])*

Recently an effective block approach has been proposed to address the experimentally-observed growth rates of fatigue cracks at critical locations on F/A-18 airframes. In this approach, each program of spectrum load is treated as an equivalent constant amplitude cycle, and the baseline crack growth rate data are obtained using a similar spectrum load of interest. A procedure was devised to allow the use of the model parameters obtained under one load spectrum to predict the crack growth under a different load spectrum. In this study, we critically evaluate the capability of the effective block approach, using data obtained for the F-111 and P-3C coupon test programs, to gauge its general applicability to other aircraft operated by the Royal Australian Air Force. The data used in the evaluation encompasses different load spectra, different materials and different crack configurations. This investigation has found that the effective block approach was able to model fatigue crack growth in 2024-T851 aluminium under the F-111 flight spectra examined (see Figure 34), but it could not produce an acceptable estimation of the total crack growth life for the P-3C spectra studied, as shown in Figure 35. It was, however, able to produce reasonable predictions of fatigue crack growth in a chosen interval of crack length. This report provides an independent evaluation and guidance for the application of the effective block approach.

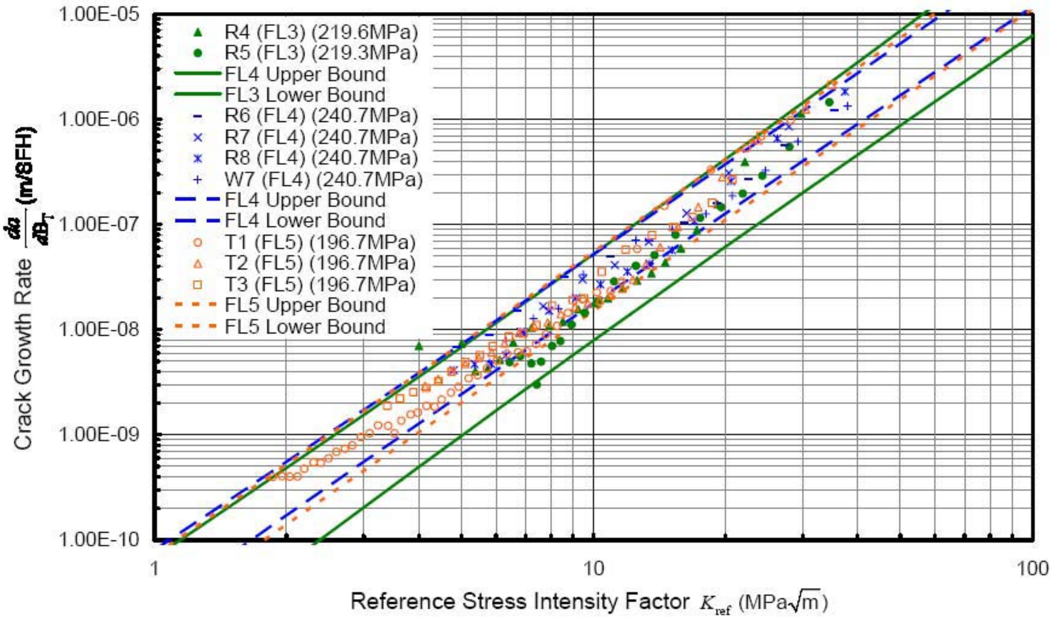

Figure 34 Prediction intervals (upper and lower limits) for FL3, Fl4 and FL5 crack growth rate data sets.

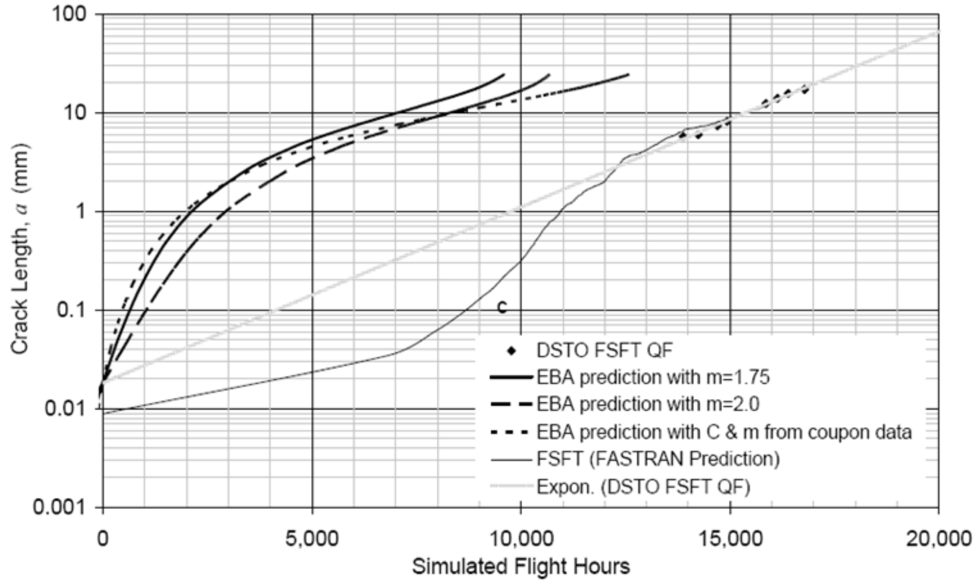

Figure 35 EBA predictions of a crack with an initial flaw size of 0.018 mm located at WS 220

References:

1. Wallbrink, C., R. Amaratunga, W. Hu, P. Jackson and D. Mongru (2008). An evaluation of the effective block approach using P-3 and F-111 crack growth data, Defence Science and Technology Organisation. DSTO-TR-2195.

2. Wallbrink, C. and W. Hu (2008). "An evaluation of the effective block approach for predicting crack growth under an untested spectrum using P-3C and F-111 test data." Advanced Materials Research **41-42**: 189-197.

8.4.5 Crack Growth Analysis Program, update on developments. *(W Hu, [DSTO])*

CGAP is MS Windows-based, and it provides a database capability for the management of geometry, material, load and solution information. A screen view is shown in Figure 36. Apart from the deterministic crack growth, it also provides a capability in probabilistic crack growth analysis based on the Monte Carlo method, with randomized initial crack size, crack growth rate and the peak spectrum stress. In addition, CGAP also interfaces seamlessly with FASTRAN 3.8.

CGAP 1.7 is about to be released. The major change is the integration of a strain-based fatigue life analysis tool. A new material database was implemented to allow the sharing of data between crack initiation and crack growth analyses. This work was conducted to support the C-130 fatigue test interpretation, but its application can be more general.

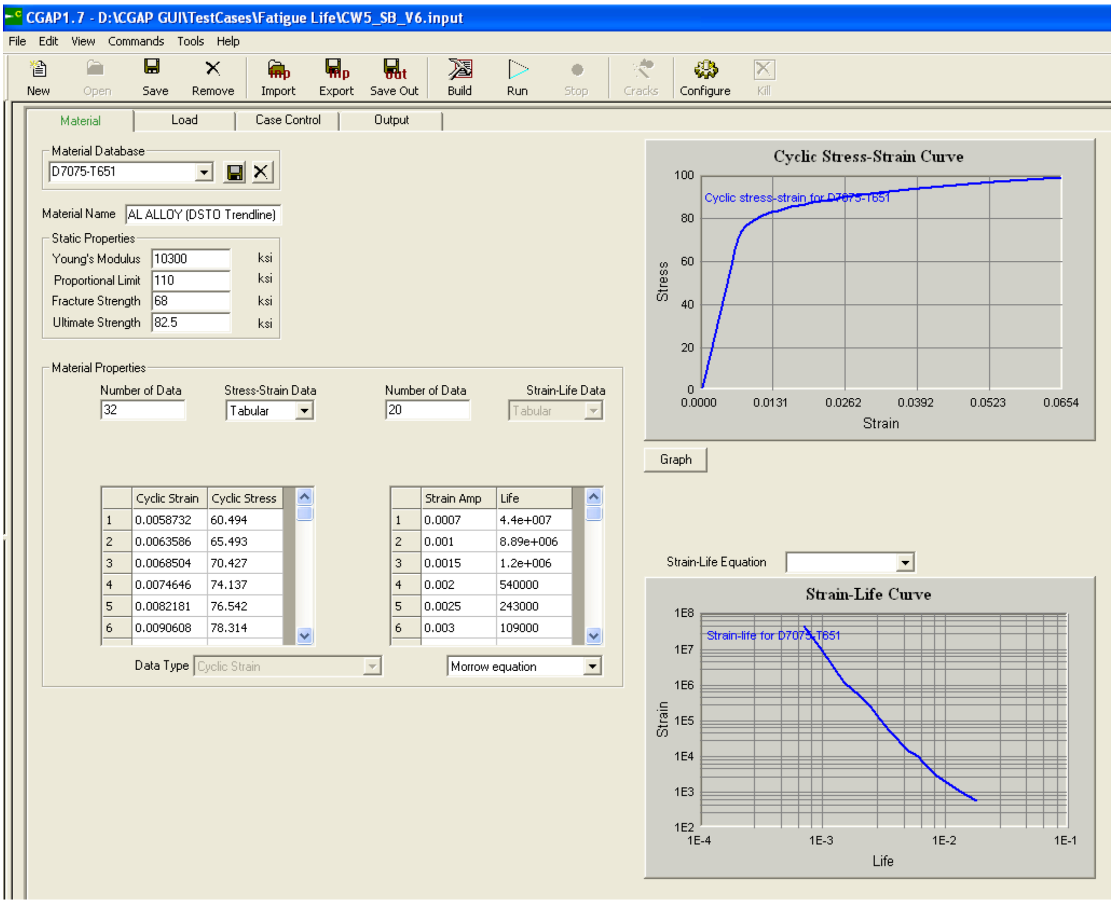

Figure 36 CGAP screen layout

8.4.6 Experimental validation of stress intensity factor solutions for the pin loaded lug *(SQNLDR D Child [RAAF ASI-DGTA] and SQNLDR N Moyle [RAAF TFLMU])*

Paper to be presented at ICAF 2009 Symposium

Abstract

Predicting crack growth behaviour is an important element of managing the structural integrity of aircraft fleets. It is common for fleet managers to employ software packages, such as AFGROW, to automate the prediction of crack growth behaviour. AFGROW has traditionally used empirically-derived stress intensity factor solutions. However, stress intensity factor solutions generated using Finite Element Models (FEMs) are now being employed within AFGROW to provide a greater level of accuracy and flexibility.

Irrespective of the move towards using FEMs, stress intensity factor solutions must still be appropriately verified in order for the results to be utilised in structural integrity management. Two independent research programs were conducted at Purdue University to review stress intensity factor solutions for cracked, pin-loaded lugs – a commonly used configuration in aircraft primary structure. The purpose of the research was to experimentally validate stress intensity factor solutions derived from FEMs

Component fatigue tests were conducted on corner, oblique and through cracked, pin loaded aluminium lugs for a range of lug geometries typical in aerospace applications. Crack propagation was measured using direct optical and marker banding techniques. Non-dimensional stress intensity factor (geometry factor) solutions were calculated utilising a back-tracking method and compared with results from representative StressCheck® Finite Element Models

In the case of the through crack configuration, the comparison of experimental and StressCheck® derived geometry factors showed a close correlation and were an improvement to solutions provided in AFGROW at the time. Based on the results of this research, StressCheck® pin-loaded lug geometry factors have since been incorporated into the AFGROW software.

In the case of the corner crack configuration, there was a correlation between experimental and StressCheck® results. However, this was highly dependant on the choice of pin-loading boundary conditions in the FEM. The results of this research have been utilised to define the default AFGROW software pin-loaded lug boundary condition assumptions.

8.4.7 Airframe life extension by optimised shape reworking - Australian experience and recent developments

(M Heller, M Burchill, R Wescott, W Waldman, R Kaye, R Evans, and M McDonald [DSTO])

Paper to be presented at ICAF 2009 Symposium

Abstract

Typically during the life of an airframe, a few key stress-concentrating locations can become fatigue critical, and an effective repair option is needed. Such repairs can provide significant economic benefits, by avoiding the need for component replacement, as well as usually increasing the interval between costly periodic in-service inspections.

The Defence Science and Technology Organisation Australia (DSTO) has developed a unique life extension approach, where optimised rework shapes are designed and used. Two distinct scenarios for optimal shape reworking are: (i) a repair, where damaged material is removed and (ii) as a pre-emptive repair measure to avoid cracking. In both cases, apart from removing material, the optimised shapes provide significantly reduced stress peaks, thus leading to increased fatigue life, inspection intervals and airframe availability.

The optimisation approach is based on an analogy with biological growth, and is implemented as an iterative gradientless finite element procedure. The optimal shapes are unique and depend on local loading and practical geometric constraints. An achievement of the DSTO work has been the transition from theory and generic developments to practical applications. Here, lessons learnt from the practical experience have also been used to enhance the theoretical methods, leading to an improved practical capability.

Hence in the present paper, we cover the key issues relating to the theoretical and practical developments for rework shape optimisation, as follows:

i) Implementation of fully automated numerical methods, in 2D or quasi-3D. Advanced features include; minimising the magnitude of the multiple, constant-stress segments around the stress concentrator boundary (i.e. holes); robustness to account for perturbations in the direction of the dominant loading, or multiple load cases; and geometric constraints such as minimum radius for manufacture.
ii) Key results from generic studies and benchmark analysis, covering both unique and transferable shapes dependent on loading and geometric conditions.
iii) Practical applications and demonstrators including lessons learnt. This focuses on open holes and section runouts, including applications for F-111 fleet aircraft, F-111 fatigue test articles, and F/A-18 and P-3C components.

iv) Fatigue life effects due to the reductions in peak stresses and stress intensity factors, relating to the relevant airworthiness philosophy, i.e. safe life or safety by inspection. Taking into account the effect on fatigue life of the following; minimised peak stress, manufacturing constraints, robustness constraints, and non-destructive inspection limits.
v) Post processing the initial numerical shapes designs for effective computer aided manufacture.
vi) Improved manufacture of optimised shapes, particularly focussing on simplified in-situ manufacture of shapes using compact jigs and semi-automated tooling.
vii) Approach to transitioning and certification of such repairs, and how optimised shapes themselves can be simply repaired should re-cracking occur (although unlikely). This includes recommendations on the effective use of the technology based on experience gained.
viii) Relevance of the technology to initial design, with an example from a combat aircraft.

In summary, this paper presents the key achievements from an extensive work program undertaken by DSTO under the sponsorship of the Royal Australian Air force over the last twelve years. Highlights include: enhancements to the optimisation method, development of novel in-situ machining techniques and the successful implementation for RAAF aircraft. As indicated above, the paper is very much in tune with the conference philosophy of 'bridging the gap between theory and operational practice'.

8.4.8 Transforming Stress Optimal Free Form Shapes for Improved Numerically Controlled Manufacture
(R Wescott, [QinetiQ AeroStructures] and M Heller, [DSTO])

Improving the specification of optimal rework shapes is important for the wider application of DSTO shape optimisation technology to RAAF airframe life extension. The research report [1] presents the development of an algorithm to smooth the variation in radius of curvature of optimal shapes that are represented by points output by finite element based optimisation codes. The equations for smoothing are based on iteratively adjusting the radius of curvature of multiple circular arcs joining the points. Figure 37 shows a robust optimal hole[1] in a large plate subject to a uniaxial stress with an angular variation of +/- 20 degrees. For an aspect ratio (a/b) of 2.1, the robust optimal hole in Figure 37 gives a peak stress concentration factor (K_t) of 2.162. This is 28 % less than a circular hole (K_t of 3) and 10 % less than an ellipse with the same aspect ratio. Figure 38(a) shows the variation in radius of curvature of the optimal shape based on points output by the optimisation code. The variation in radius of curvature after ten iterations of the smoothing algorithm is shown in Figure 38(b). The shape of the smoothed optimal hole is virtually indistinguishable from the original, and predicted K_t increases by a negligible amount to 2.180.

Derivations of equations are also presented to create intermediate optimal shapes and approximate optimal shapes composed of only a few arcs [1]. For shape morphing, intermediate optimal shapes are generated by interpolating between two bounding optimal shapes. A weighted average radius approach is used to determine approximate optimal shapes composed of only a few arcs. The abovementioned numerical techniques have been implemented in FORTRAN computer programs. In all cases the shape for manufacture is represented by circular arcs defined in IGES file format. Demonstration cases are presented that include generic robust holes, F-111 aircraft hot spot geometries and a manufactured test specimen. The results show that optimal shapes can be successfully transformed using the abovementioned methods without degrading their optimality, and that the process for transferring free form shapes leads to effective manufacture.

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[1] The peak stress around a *robust optimal* hole is minimal and the same for all load angles in the stated range.

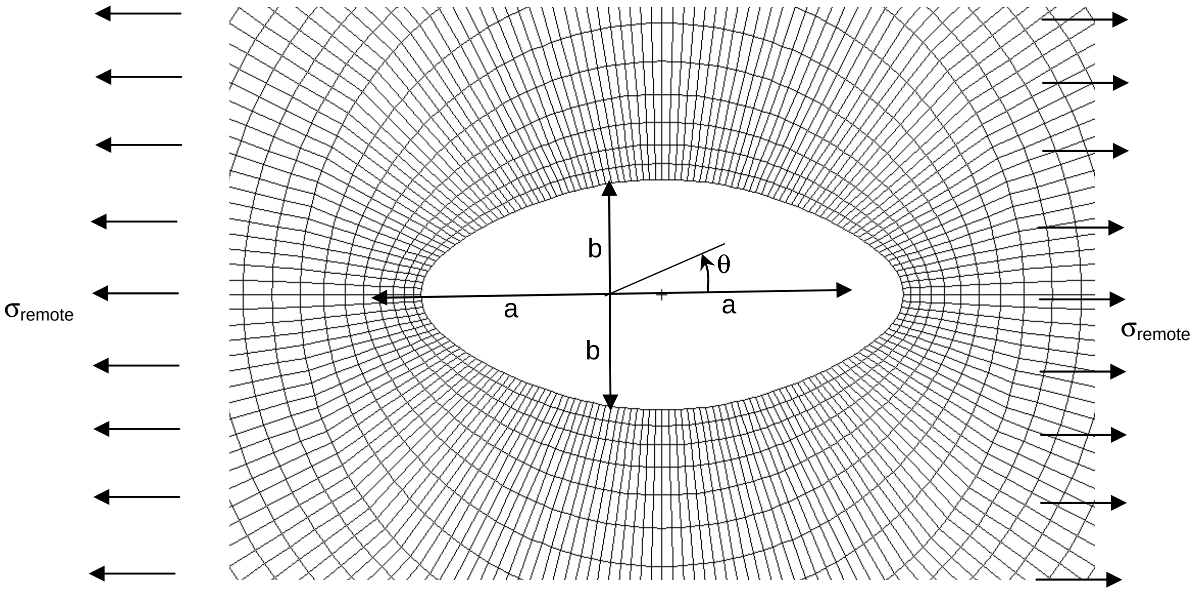
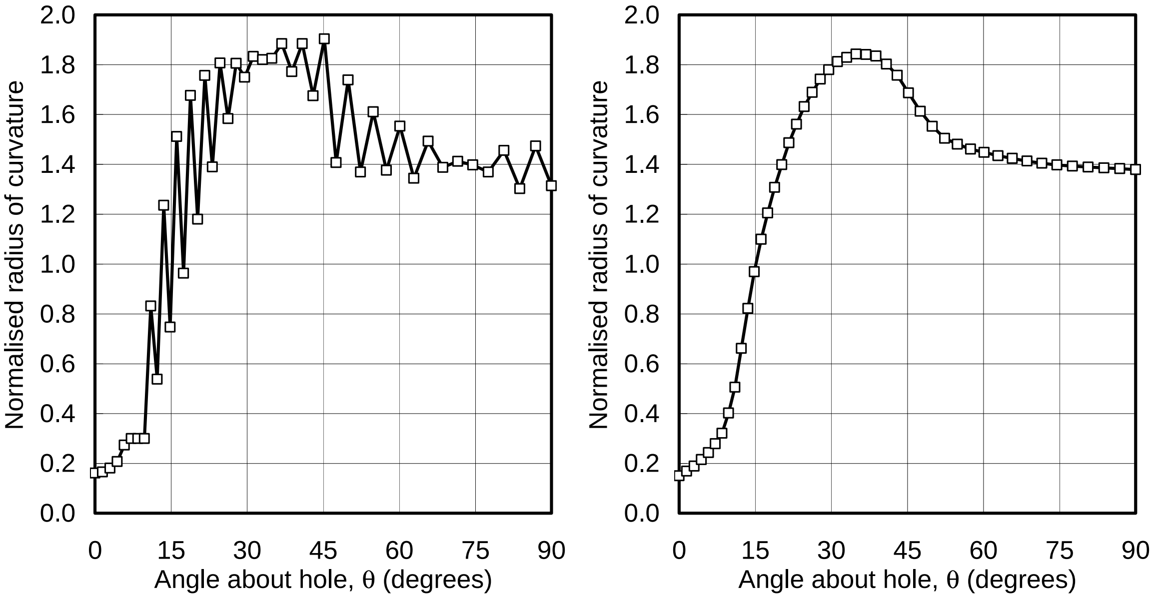


Figure 37 Problem definition and local detail of FE mesh used for robust optimal shape.



(a) No smoothing *(b) 10 smoothing iterations*

Figure 38 Normalised radius of curvature of an optimal shape - no smoothing and 10 smoothing iterations.

Reference

1. Wescott R and Heller M, Transforming stress optimal free form shapes for improved numerically controlled manufacture, DSTO Research Report, in publication, March 2009.

8.4.9 Stress and Fracture Analysis of Circular Arc Blends for Repair of Cracked Metallic Components *(M Burchill and M Heller [DSTO])*

This section presents compact functions for peak stresses and stress decay distributions due to circular arc blend repairs, on circular holes in uniaxially loaded plates, [1]. Key geometric parameters are varied, focusing on relatively shallow blends, consistent with the in-service repair of fatigue cracking in ageing airframes. The highly accurate stress results are obtained using adaptive p–version finite element analyses. In the 2D cases, both single and double sided repairs are considered, where the blend consists of either a single or multi-arcs. For the 3D analyses, the focus is on blends to repair corner cracks at the hole edge. It is found that in all cases, the peak stress can be significantly reduced by maximising the ratio of blend radius to hole radius, thereby lowering the possibility of re-cracking. The multi-arc case offers the greatest stress reduction. The stress decay functions presented can be used to obtain stress intensity factors for the fatigue analysis of cracks which may re-initiate at the blends.

The typical 2D and 3D blend geometries considered are shown below in Figure 39 and Figure 40 respectively. As an example, many parametric results were obtained for the single circular arc blend (i.e. with $r_2 < r_1$) as a function of d/r_2 and r_2/r_1 . An expression for K_t can be determined using linear regression, by fitting to the linear region as:

$$K_t = 0.440n\frac{d}{r_2} - 1.300\frac{r_2}{r_1} + 4.340 \quad \text{where } n \text{ is the number of blends, (i.e. 1 or 2)} \qquad (1)$$

Use of this equation gives K_t values to within 1% of the initial FEA results, in the range: $0.05 \le d/r_1 \le 0.50$ and $0.7 \le r_2/r_1 \le 1.0$. As expected the lowest K_t as a function of d/r_1 is given by:

$$K_t = 3.040 + 0.440n\frac{d}{r_2} \qquad \text{when} \qquad r_2 = r_1 \qquad (2)$$

Further compact expressions for other 2D and 3D cases, for peak stress, stress decay and stress intensity factor are given in Ref. [1].

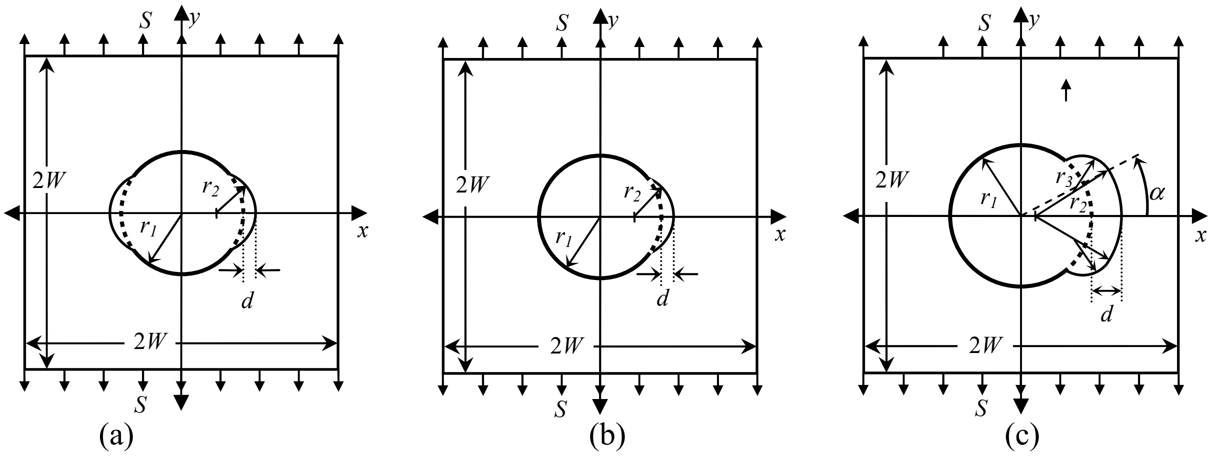


Figure 39 Geometry and notation for large 2D plates with a circular hole and circular arc repairs: (a) one circular arc blend on both sides of hole, (b) one circular arc blend on one side of hole and (c) multi-arc blend on one side of hole.

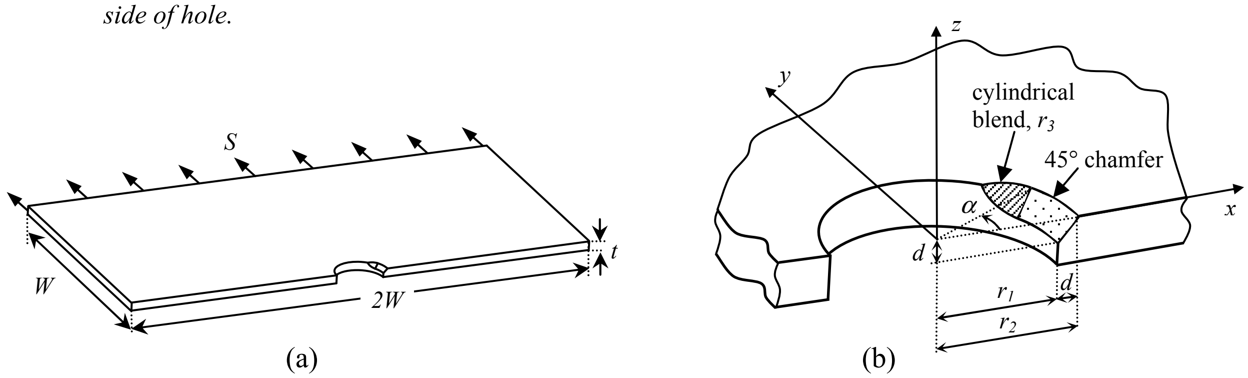


Figure 40 Geometry for hole in a large 3D plate with multi-arc blend on one side of hole: (a) plate dimensions and loading, with only ½ section shown, and (b) local details around blend.

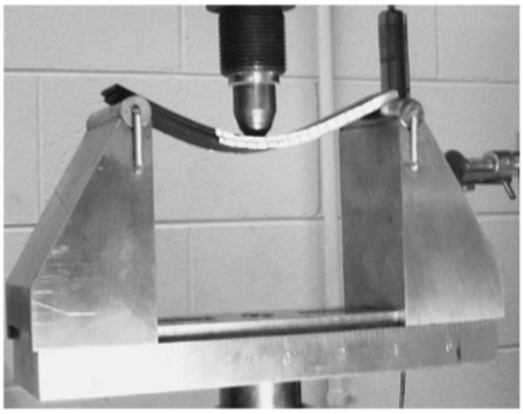
Reference

1. Burchill M and Heller M, Stress and Fracture analysis of circular arc blends for repair of cracked metallic components, *5th Australasian Congress on applied Mechanics, ACAM 2007, 10-12 December 2007,* Brisbane, Australia.

8.4.10 Characterisation of bonded joints under elevated strain rates *(M Scott [CRC-ACS])*

The Cooperative Research Centre for Advanced Composite Structures Ltd (CRC-ACS) has been characterising mechanically fastened and adhesively bonded joints in composite structures at elevated strain rate in collaboration with the German Aerospace Center (DLR), RMIT University, University of New South Wales (UNSW) and Pacific Engineering Systems International Pty Ltd (Pacific ESI) using both experimental and theoretical investigations. In recent research, toughened epoxy adhesive joints made using FM300-2, typically used in aerospace structures, were compared to joints made using the patented Thermoset Composite Welding (TCW) process where the thermoset substrates are welded together using a thermoplastic film.

The experiments included both mixed mode and pure Mode II tests with non-constant strain rates ranging from 0 to 80/s achieved using a modified drop tower as per Figure 41, with the crack propagation measured using high speed video at 16,000 fps.



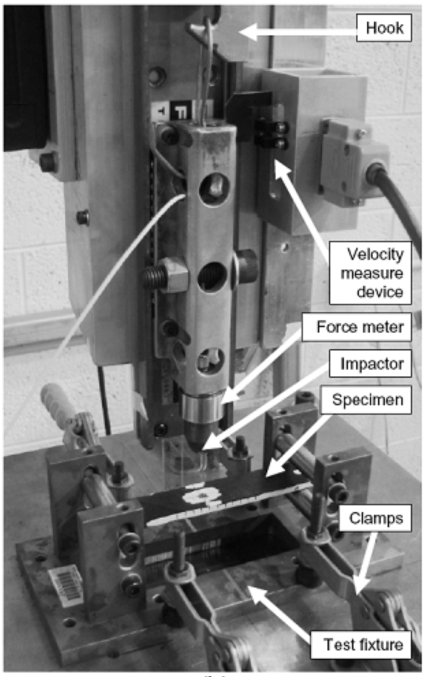


Figure 41 Test set-up for (a) quasi-static (mode II shown) and (b) dynamic characterisation of bonded joints

It was concluded that the TCW joints were stronger and less sensitive to strain rate effects than the toughened epoxy adhesive joint. Fracture for both types of joints was dominated by a “stick-slip” behaviour resulting in crack propagations of a highly variable nature. The theoretical investigations were conducted using Linear Elastic Fracture Mechanics (LEFM) and Cohesive Zone Modelling (CZM). Future work will include strain rate sensitive modelling and experimental improvements in crack energy losses and transient crack data measurements.

8.4.11 Fatigue crack growth path studies *(P White [DSTO])*

The fatigue mechanisms group, part of the Air Vehicles Division of DSTO, has been involved in studying the mechanisms behind fatigue crack growth. Focus has been on AA7050 material which is used in existing and new fighter aircraft. Recent investigations have shown how the formation of marker bands on the fracture surface is related to the mechanism of striation formation. One member of the group, Dr. Simon Barter has spent 2009 on attachment at NLR in the Netherlands working with Dr. Russell Wanhill in furthering the understanding of marker band formation in aircraft materials [1]. Marker bands can be produced with different combinations of loading which result in changes to the crack growth path which gives a visible pattern.

Marker bands not only provide understanding of the mechanism of striation formation [2] but also provide a way to measure bands of constant amplitude growth that otherwise cannot be directly measured. Marker bands produced by periodic underloads, Figure 42, have been used in this way to mark small segments of constant amplitude cycles to determine the rate of crack growth of small cracks starting from material defects. These techniques provides direct measurement of crack growth rates that can be used for variable amplitude predictions.[3]

To aid in the understanding of the mechanisms a finite element approach has been investigated with the aim of modelling the behaviour of striation formation. This approach used a newly developed cohesive zone model which used two different elements; one element to model the base material properties and another to model the damage and separation behaviour. However, this approach had difficulty in converging to a solution, and further work is continuing, see section 8.4.12.

Another approach being investigated is the use of atomic simulation to model the dynamic behaviour of atoms at the crack tip. Thus far this is showing some promise in modelling pure aluminium exhibiting some of the behaviour previously postulated in order to explain the creation of marker bands, such as growth through shear bands, separation along the interface between slipped and unslipped material and asymmetric crack tip formation, Figure 43.

This work continues at DSTO where the simulations will be increased in sophistication, new materials will be investigated and new techniques will be used to further investigate the crack growth process. Further work on marker band formation is also the subject of PhD studies at the TU Delft, The Netherlands by Milan Krkoska.

1. Barter, S.A and Wanhill, R.J.H., Marker Loads for Quantitative Fractography (QF) of Fatigue in Aerospace Alloys, National Aerospace Laboratory NLR, NL, NLR-TR-2008-644, November 2008.

2. White P.D, Barter, S.A, Molent L, Observations of crack path changes caused by periodic underloads in AA7050-T7451, International Journal of Fatigue 30 (2008) 1267--1278.

3. White, P., Barter, S.A. and Wright, C., Small crack growth rates from simple sequences containing underloads in AA7050-T7451 International Journal of Fatigue, http://dx.doi.org/10.1016/j.ijfatigue.2009.01.014

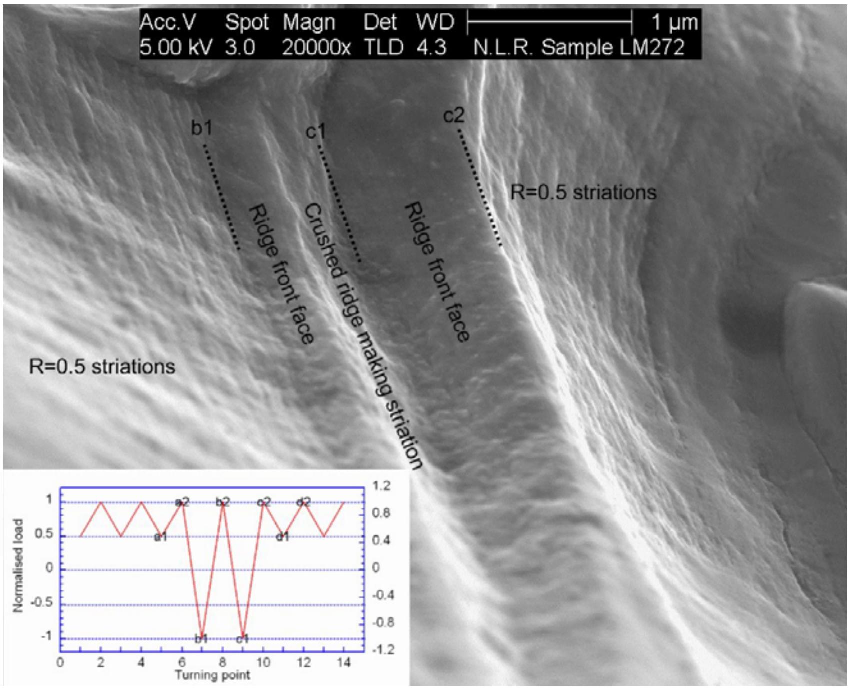


Figure 42 Striation and ridge formed by the application of two underloads in a sequence of R=0.5 loads. The striation appears to be a ridge (produced by the load segment from b1 to b2) that has been partially crushed by the load segment from b2 to c1 in the schematic. [3]

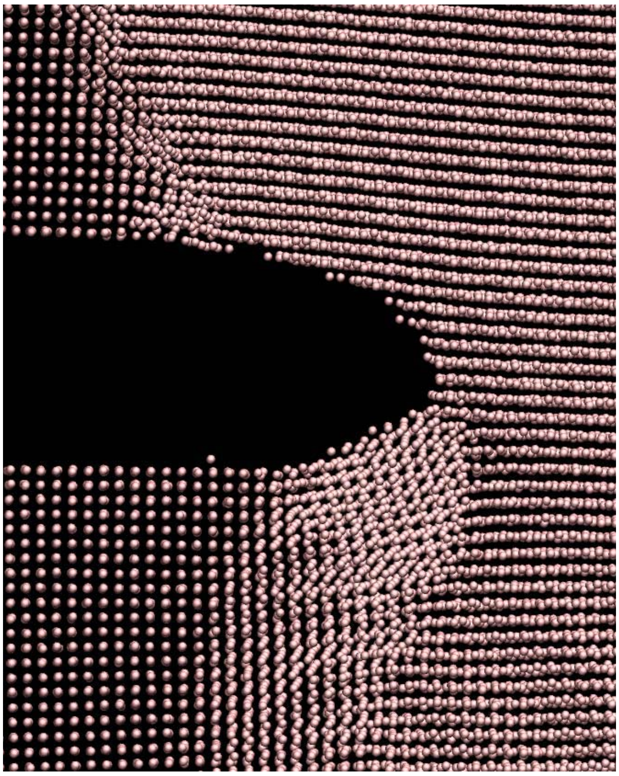


Figure 43 Atomic simulation of a short crack tip for Al using embedded atom potential (EAM) orientation (100). Model consists of 453005 atoms with simulated Mode I loading.

8.4.12 A Slip Band Based Cohesive Zone Model for Simulation of Fatigue Crack Growth at Sub-micron Level *(X Yu, C Wright and M Heller [DSTO])*

The purpose of this study [1] was to gain further understanding of fatigue crack growth mechanisms at micro-level. It was motivated by recent observations [2] of fatigue crack surfaces in AA7050-T7451 alloys, which were produced by load sequences consisting of periodic underloads (R=−1) in between groups of high stress ratio (R = 0.5) load cycles. Ref [2] suggested that fatigue crack growth in this material was due to formation of coarse slip bands followed by slip band cracking. A slip band based cohesive zone model was thus proposed, assuming that: (i) plasticity was limited to slip planes and elasticity was retained elsewhere; (ii) tangential resistance of a slip band followed the rule of kinematic and isotropic cyclic hardening; (iii) normal strength of a slip band degraded when cyclic sliding accumulated; and (iv) cracking occurred when resolved normal stress reached locally degraded strength of the slip band. The model is illustrated in Figure 44.

The cohesive zone model was implemented in ABAQUS using UEL and UEXTERNALDB user subroutines. Case studies were performed on an edge-cracked plate under remote cyclic tension, see Ref [1] for model descriptions. Simulation results showing progressive and branched crack growth were given in Ref [1], *see* Figure 45.

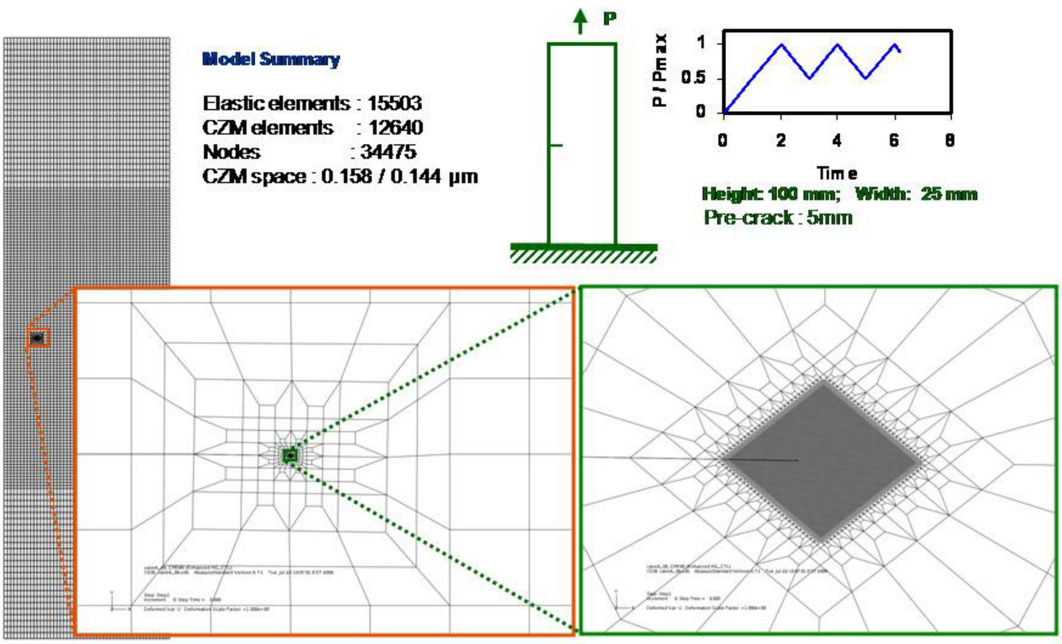
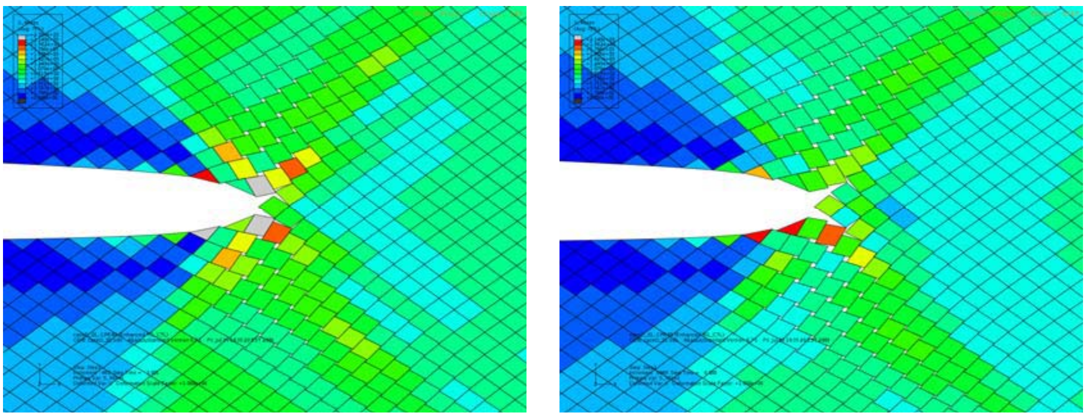


Figure 44 Finite element model for case study.



(a) t = 2.0 second (b) t = 6.0 second

Figure 45: Progressive growth of fatigue crack as predicted by the cohesive zone modelling.

References

1. Yu X, Wright C, Heller M. A slip band based cohesive zone model for simulations of fatigue crack growth at sub-micron level, presented at: XXII ICTAM, 25-29 August 2008, Adelaide, Australia.
2. White P, Barter S, Molent L. Observations of crack path changes caused by periodic underloads in AA7050-T7451, *International Journal of Fatigue*, 30, pp 1267-78, 2008.

8.4.13 Remote synthesis of loads on helicopter rotating components using linear regression, load path and statistical analyses, *(X Yu, C Wright and M Heller [DSTO])*

Maintenance costs of helicopters can be reduced by applying individual tracking of rotating component usage. This study [1] investigated the possibility of synthesizing a load-time history of the main rotor pitch link (MRPR) from strain measurements on stationary components below swashplate. Four approaches were tried, namely linear regression in the frequency domain, linear regression in the azimuth domain, load path analysis of the swashplate assembly, and supplementary statistical analysis. Cross-validation and an advanced linear regression method called Elastic Net were employed. All the approaches of load syntheses were tested against a data set containing 42 runs of level flights selected from the Joint USAF-ADF S-70A-9 Black Hawk flight strain survey conducted in 2000. A typical comparison of synthesized and measured load-time histories is shown in *Figure 46* for two flight conditions. The main finding of the study was that an accurate or complete load synthesis was not always possible due to the existence of reactionless force components, see *Figure 47*

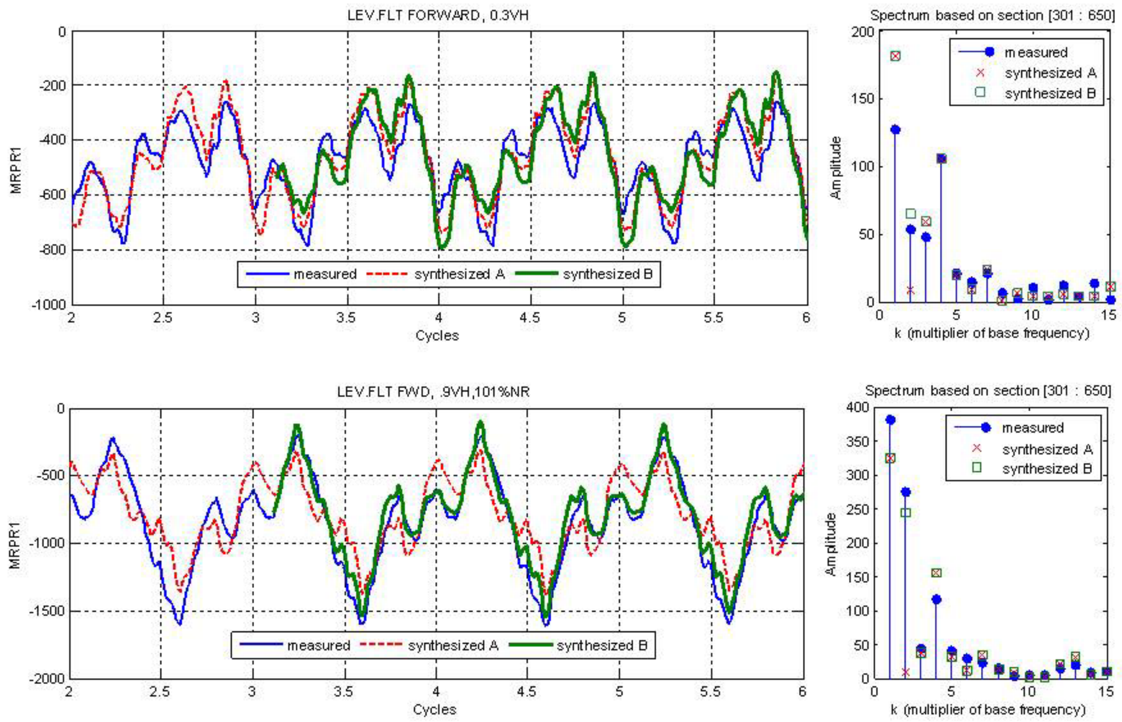


Figure 46 A comparison of measured and synthesized loads on MRPR1 under two level flight conditions. (a) 0.3Vh: synthesized data do not match well with measurements; (b) 0.9 Vh: synthesized B reasonably matches the measured data . ("synthesized A" was obtained using load path analysis. "synthesized B" was obtained using load path analysis then supplemented by statistical analysis.)

This remained true no matter what synthesis approach was applied. The load path analysis was recommended as an essential step in remote synthesis of loads on helicopter rotating components. This recommendation applies to activities including evaluating reported case studies, utilizing existing data sets, and developing new strategies of load synthesis.

A follow-up study is being carried out to clarify whether or not the strains on stationary swashplate provide sufficient fidelity to synthesize load history on MRPR.

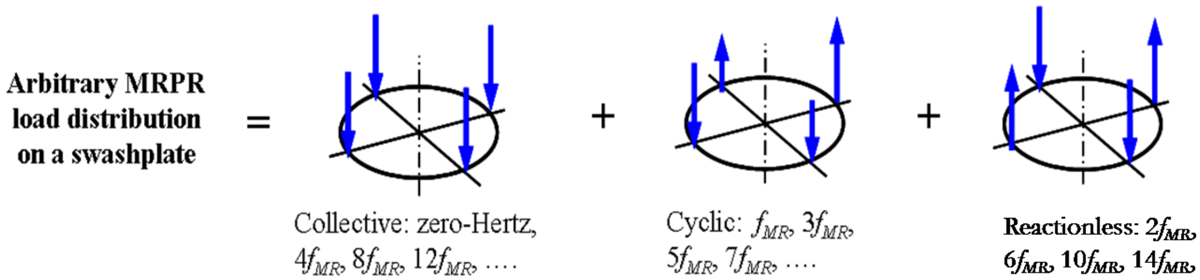


Figure 47 Any arbitrary MRPR load distribution on a swashplate can be expressed as a combination of collective, cyclic and reactionless components. The reactionless component is self-balanced on the swashplate and cannot be detected down stream the load path below the swashplate.

Reference

1. Yu X, Wright C, Heller M. Remote synthesis of loads on helicopter rotating components using linear regression, load path and statistical analyses, presented at: 34th European Rotorcraft Forum, 16-18 September 2008, Liverpool, UK.

8.4.14 Comparison of fatigue crack growth from three different stress concentrations *(J Huynh, S Barter and L Molent [DSTO])*

Fatigue crack growth has been observed by many investigators to be influenced significantly by mechanical details such as notches, from which fatigue cracking initiate. The severity of these notches is usually accounted for by the stress concentration factor (K_t). Although the effects of the higher K_t*'s* are known to result in a reduction in fatigue life compared to low K_t details, there appears to be little work reported in the literature that directly compares experimental crack growth data for different K_t*'s* under either constant amplitude or variable amplitude loading.

An investigation of short fatigue crack growth in 7050 aluminium alloy from details with different K_t*'s* was undertaken [1, 2], so that empirical crack growth models could be developed that are capable of accounting for different K_t*'s*, and at different applied stress levels. The objective of this research was to develop a useful tool in predicting crack growth of fatigue cracks emanating from different stress concentration features in the F/A-18 structure.

The task involved performing fatigue tests on three sets of coupons made of the same material, each with a different K_t (*Figure 48*). These coupons were subjected to the same loading spectrum, but were tested at different applied peak stress levels. The material used was 7050-T7451 high strength aluminium alloy, while the spectrum used was a representative F/A-18 wing root bending moment spectrum. The spectrum had additional marker loads added to aid Quantitative Fractography (QF).

The purpose of maintaining the same material and loading spectrum, while varying the coupon K_t and applied stress, was to isolate the effect of the K_t on the crack growth rate, and to aid the subsequent model development. In each of these fatigue tests, cracks were grown naturally, with no crack starters applied. The crack growth rate analyses were based on QF results obtained from the lead cracks in each of the coupons tested. The Effective Block Approach (EBA) for crack growth modelling was used [3], so that any spectrum effects, retardation or crack closure mechanism would be accounted for within each spectrum block of crack growth measurement.

The equivalency of two traditional empirical crack growth models, namely the Paris and Frost & Dugdale models, when the apparent stress dependency in the Paris-like C constant was accounted for [2].

It was also observed that the stress intensity factor K, when plotted against the crack growth rate per block, did not appear to collapse the data from the three coupon sets (Figure 49). Here

$$K_{ref} = \sigma_{ref} \beta \sqrt{\pi a}$$

where σ_{ref} is an arbitrary reference stress from the block spectrum, and β is a function accounting for the changing stress intensity factor due to the changing crack front shape, component geometry, and remote loading constraint conditions. As the peak spectra stress was used throughout the analysis, the stress intensity considered was therefore K_{max}.

The crack growth models developed were shown to successfully relate higher K_t coupon results to low K_t short crack growth data [1, 2], or vice-versa, and hence appears to be a useful tool to support the lifing of the F/A-18 structure.

References:

1. Huynh J, Molent L and Barter S. Experimentally Derived Crack Growth Models for Different Stress Concentration Factors; J. Fatigue, Vol 30/10-11 pp 1766-1786.
2. Huynh J, Molent L and Barter S, Fatigue crack growth predictions for 7050 Aluminium Alloy with Different Stress Concentration Factors, DSTO-RR-0330, Jul 2007.
3. Molent, L., McDonald, M., Barter, S., Jones, R., Evaluation of Spectrum Fatigue Crack Growth using Variable Amplitude Data, International Journal of Fatigue (2007); 30/1; 119-137.

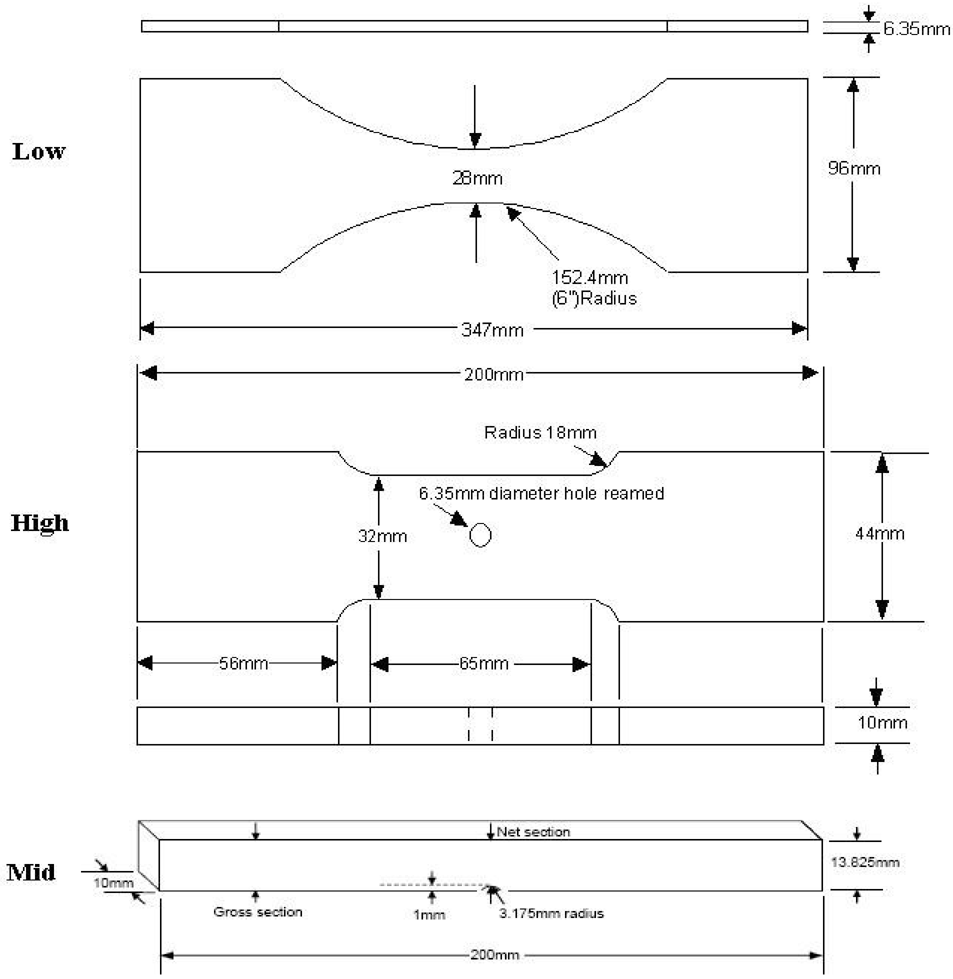


Figure 48 Diagram of the three coupon types.

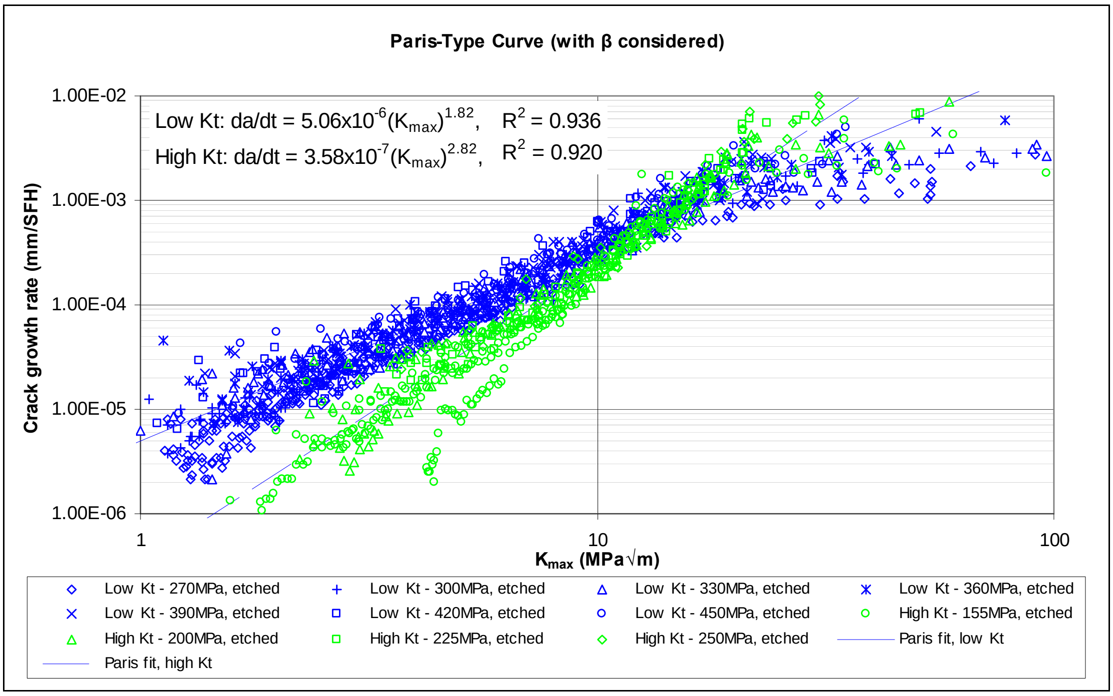


Figure 49 Paris-type curves showing the relationship between the $K_{t(low)}$ and $K_{t(high)}$ CG rate data when β (from AFGROW) was considered.

8.4.15 Application of FAA-Proposed Rules for Preclusion of Widespread Fatigue Damage *(P Jackson and D Mongru [DSTO])*

US Federal Aviation Regulations Part 25.571 for transport aircraft require a damage tolerance and fatigue evaluation to be carried out on the aircraft's structure. In 1998 amendment 25-96 introduced the requirement for test based clearance against Widespread Fatigue Damage (WFD) and stated that no airplane may be operated beyond half the equivalent number of cycles accumulated on the fatigue test article, see Figure 50. More recently, the Regulation and its accompanying Advisory Circular AC25.571-1C are proposed to be updated to provide more specific instructions regarding the preclusion of WFD in aircraft structures, including the establishment of an Initial Operating Limit (IOL).

From 1999 to 2004, as part of the international P-3 aircraft Service Life Extension Program (P-3 SLAP), the wing and fuselage of a retired airframe was subjected to a full scale fatigue test conducted by Lockheed-Martin. The article was cycled to beyond two lifetimes and considerable damage was accrued, including a number of examples of WFD, both as Multi-site Damage (MSD) and Multi-element Damage (MED). Subsequent to the test and its accompanying teardown, the Australian Defence Science and Technology Organisation (DSTO) conducted a test interpretation program based on operating the aircraft under the FAA 25.571 Damage Tolerance Analysis requirements. The DSTO work not only included the traditional damage tolerance and fatigue life analyses, but also an additional analysis that used as its methodology the recent advice in the draft update to AC 25.571-1C.

To DSTO's knowledge this was the first time that the most recent FAR 25.571 rules and guidance regarding WFD related structural limits have been tested in practice. The availability of statistical data proved to be a key part of the interpretation, particularly for the definition of "WFD Average behaviour". Fortunately the amount of damage accrued by the P-3 test wing and subsequently collected from inspections of USN aircraft enabled this data to be generated. These data are provided in *Figure 51*. The work was presented in the DTAS 2007 conference [1] and has been used in the in-service management planning for the RAAF P-3 fleet.

Reference

1. P. Jackson, D. Mongru A, *Practical Application of the Recent FAR 25.571 Requirements and Guidance for the Preclusion of Wide Spread Fatigue in a Transport Aircraft* First International Conference on Damage Tolerance of Aircraft Structures, Delft, The Netherlands, September 2007

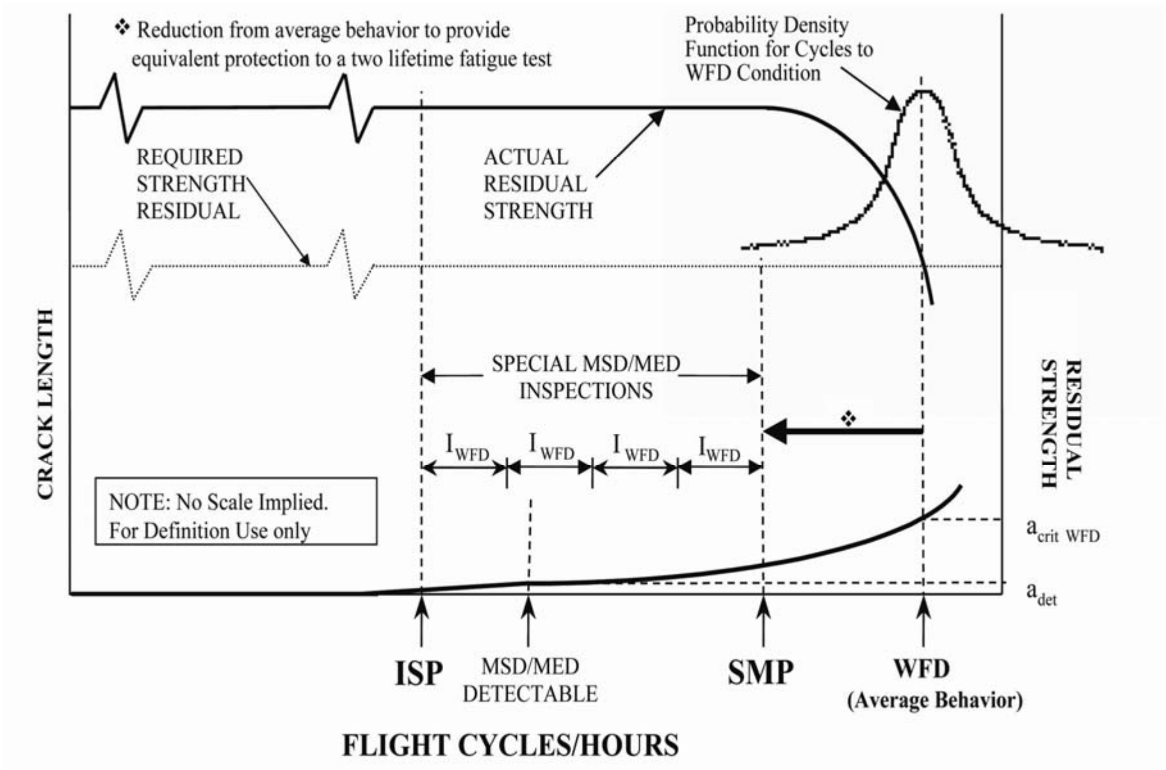


Figure 50 WFD related decision points as defined in the Advisory Circular.

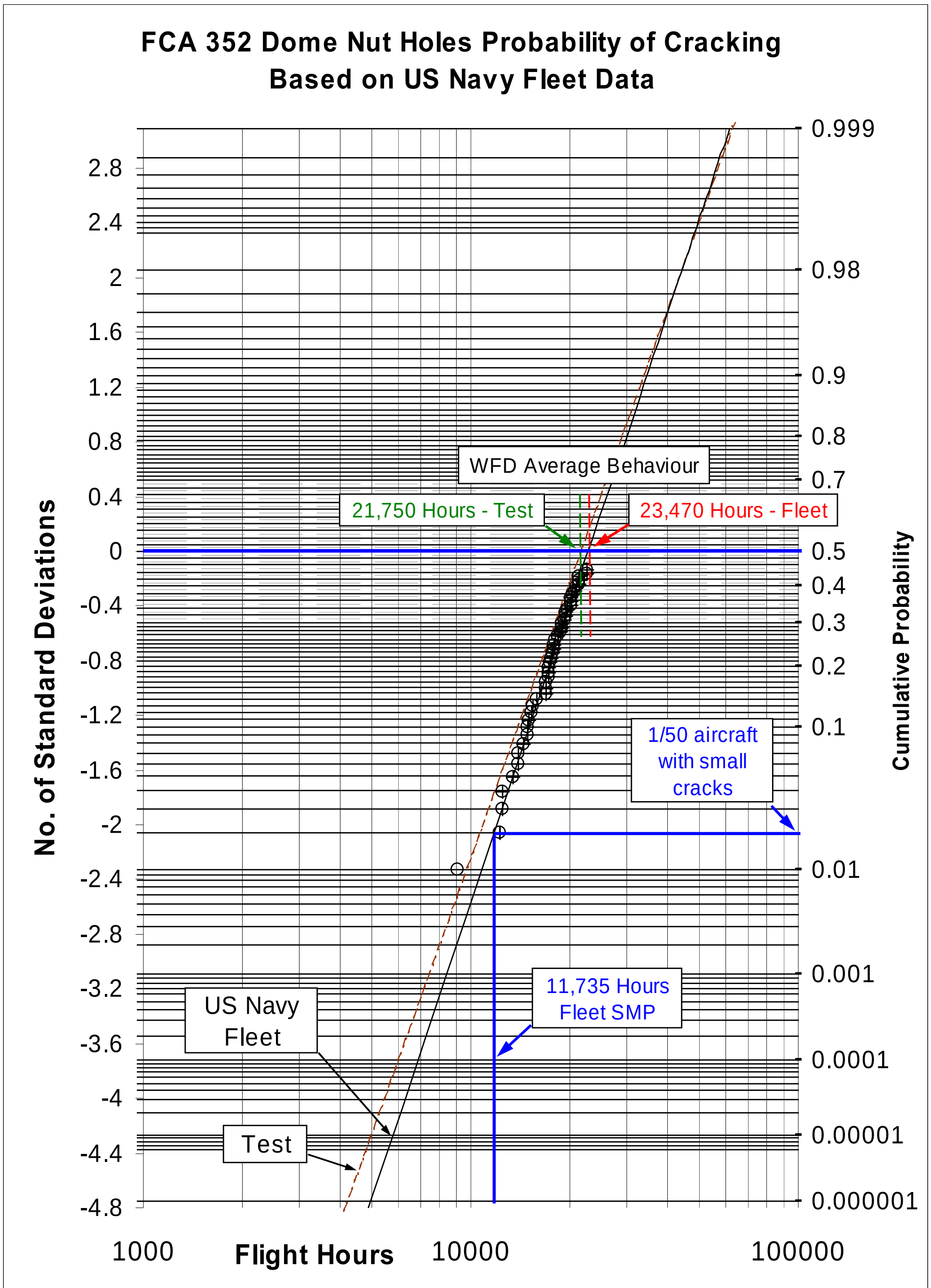


Figure 51 Plot of cumulative probability versus flight hours for USN fleet inspection findings for a critical location on the P-3 wing (FCA 352). Data enabled determination of WFD Average behaviour and Structural Modification Point (SMP) as well as comparison against the P-3 SLAP full scale test.

8.4.16 Mapping Crack Closure Using Shear Wave Ultrasound *(S Bowles, C Harding, and G Hugo [DSTO])*

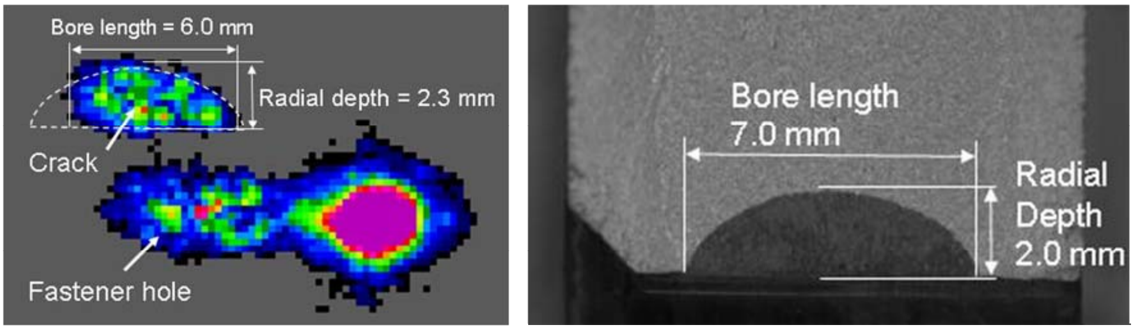
Crack closure has long been identified as a source of variability in the detection of fatigue cracks during an ultrasonic inspection [1-3]. Tightly closed cracks (those with significant contact between the crack faces due to a strong localized compressive stress acting to close the crack faces together) may give a significantly reduced ultrasonic response compared to an open crack (one with no contact between the crack faces).

A recent experimental study by DSTO examined the effect of crack closure on the ultrasonic response for fatigue cracks at fastener holes grown in aluminium alloy specimens using a high-fidelity flight-by-flight load spectrum representative of Royal Australia Air Force F-111 aircraft usage [4]. An automated ultrasonic scanning system was used to make detailed measurements of the 45° shear-wave pulse-echo ultrasonic response as a function of tensile or compressive load applied to each specimen.

Figure 52 shows the ultrasonic C-scan image obtained from a typical fatigue crack compared to the actual crack profile determined from fractography after breaking the crack open. The ultrasonic data were acquired whilst the specimen was under sufficient tensile load to fully open the crack faces. For this crack, the reflected ultrasound has been directly back-scattered from the surface of the crack and does not involve any skip reflections from the back face of the specimen; the ultrasonic reflection in the C-scan is thus a projection of the crack profile onto the plane of the C-scan by the 45° beam angle. There is good agreement between the crack profile from the ultrasonic data and that from fractography, with the exception of the top part of the crack, for which the ultrasonic reflection is obscured by the countersink.

Figure 53 shows the change in the ultrasonic C-scans due to crack closure for a further two typical cracks. The C-scans show the extent of the crack visible without any applied stress. The superimposed white outlines show the perimeter of the ultrasonic indication for the crack when the specimen is under an applied tensile stress of 125 MPa sufficient to fully open the cracks. For the smaller crack in *Figure 53(a)*, the ultrasonic indication is much smaller when the crack is closed than when it is open, whereas for the larger crack in *Figure 53(b)* the relative difference is smaller. *Figure 53(a)* shows that when there is no load on the specimen it is not only the tip of the crack that closes and becomes less detectable with ultrasound, but rather significant changes in the indication also occur at the mouth of the crack and along the bore of the hole. This is consistent with previously published results from theoretical modeling of crack opening displacement for cracks at cold-worked fastener holes [5].

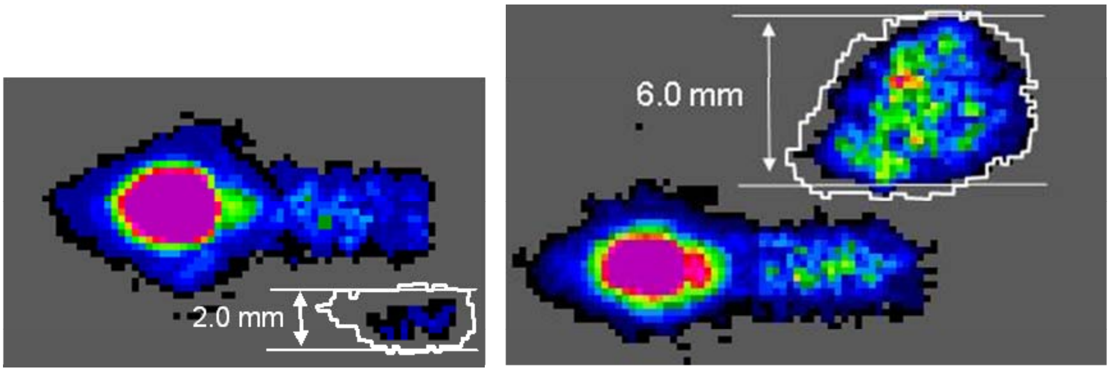
This study demonstrates the value of ultrasonic C-scan measurements on fatigue cracks as a function of applied load to investigate the nature of crack closure and its effect on the detectability of fatigue cracks.



(a) (b)

Figure 52 Comparison of (a) ultrasonic C-scan image and measured crack dimensions with (b) fractographic image and actual dimensifor a crack in a 12 mm thick specimen. Ultrasonic data were acquired using 15 MHz transducers with the specimen under tensile load to fully open the cracks. The crack profile from the fractographic image (b) is shown superimposed on the C-scan (a) by the broken curve.

Ultrasonic inspection procedures are often developed using machined notches as artificial defects in place of genuine fatigue cracks. This study shows that the true sensitivity and reliability of angle-beam ultrasound to detect *small* fatigue cracks will be significantly poorer than that assessed based solely on trials on notches. The results confirm that small cracks can be effectively undetectable using ultrasound even when comparable size notches in the same geometry give strong ultrasonic reflections. The results of this study have been used as input data to a model-assisted probability of detection (MAPOD) assessment of automated ultrasonic inspections for the lower wing skin of F-111 aircraft [6].



(a) (b)

Figure 53. Comparison of ultrasonic indications with and without applied tensile load on specimen: (a) 2 mm radial depth crack in the centre of the fastener hole bore, (b) 6 mm radial depth crack near the top of the fastener hole bore (just under the countersink). The C-scan images were acquired using 15 MHz transducers with no load on the specimen. The white outlines superimposed on the C-scans indicate the perimeters of the corresponding indications recorded with an applied stress of 125 MPa on the specimen sufficient to fully open the cracks.

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1. Thompson, R.B., B.J. Skillings, L.W. Zachary, L.W.J. Schmerr, and O. Buck, (1983) "Effects of Crack Closure on Ultrasonic Transmission" in *Review of Progress in Quantitative Nondestructive Evaluation*, D.O. Thompson and D.E. Chimenti, Editors. Vol. 2A. New York: Plenum Press. 325-343.
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4. Bowles, S.J., C.A. Harding, and G.R. Hugo, (2009 (In press.)) "Effect of Crack Closure on Ultrasonic Detection of Fatigue Cracks at Fastener Holes" in *Review of Quantitative Nondestructive Evaluation*, D.O. Thompson and D.E. Chimenti, Editors. Vol. 28. Melville, NY: American Institute of Physics.
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6. Harding, C.A., G.R. Hugo, and S.J. Bowles, (2009 (In press)) "Application of Model-Assisted POD Using a Transfer Function Approach" in *Review of Quantitative Nondestructive Evaluation*, D.O. Thompson and D.E. Chimenti, Editors. Vol. 28. Melville, NY: American Institute of Physics.

8.4.17 Inspection of a Full-Scale Fatigue Test Wing Under and Applied Load *(M Khoo, G Hugo, C Harding, S Bowles, H Morton and M Ryan [DSTO])*

DSTO and RAAF Non-Destructive Testing Standards Laboratory developed automated ultrasonic inspection procedures to detect fatigue cracking at fastener holes in the F-111 lower wing skin [1]. The procedures use a Westinghouse-AMDATA Intraspect system for ultrasonic data acquisition and motion control, linked to an automated X-Y scanner and a recirculating couplant immersion scanning head, which enables the use of focussed ultrasonic immersion transducers. The inspections are being carried out on all fleet aircraft by Boeing Australia NDT personnel.

DSTO research to support the application of automated ultrasonic testing to the F-111 wings included experiments to validate the sensitivity and reliability of the automated ultrasonic inspection procedures for fatigue cracks grown in the laboratory under representative F-111 fatigue load spectra. These experiments confirmed that the detection sensitivity for small fatigue cracks can be significantly enhanced if a tensile load is applied to the crack location to open the crack faces, which mitigates the effect of crack closure on ultrasonic response. The effects of applied tensile stress on ultrasonic detection sensitivity are most pronounced for small cracks (less than 1 to 2 mm in size) and are minimal for large cracks (e.g. exceeding 4 to 5 mm in size) [2].

The completion of fatigue cycling on the F-WELD full-scale fatigue test wing provided an opportunity to conduct an automated ultrasonic inspection of the lower wing skin with the test wing loaded to create a tensile stress in the lower wing skin. Conducting the inspection with the lower skin under tensile stress maximised the detection sensitivity for any small fatigue cracks at fastener holes (or other inspected crack-prone locations). This approach was validated by

the detection of a number of small cracks which were not readily visible in ultrasonic data obtained with the wing unloaded.

The results of the automated ultrasonic inspections provided a valuable initial assessment of the degree of fatigue cracking in the test wing and were helpful to guide the subsequent teardown activities. The inspections significantly enhanced the effectiveness of the teardown process compared to selecting a purely random sample of fastener holes for detailed teardown inspection.

1. Harding, C.A., G.R. Hugo, and S.J. Bowles, (2007) *Automated Ultrasonic Inspection for Crack Detection at F-111 Lower Wing Skin Fastener Holes.* Non-Destructive Testing Australia, **44**(3): p. 80-85.
2. Bowles, S.J., C.A. Harding, and G.R. Hugo, (2009 (In press.)) "Effect of Crack Closure on Ultrasonic Detection of Fatigue Cracks at Fastener Holes" in *Review of Quantitative Nondestructive Evaluation*, D.O. Thompson and D.E. Chimenti, Editors. Vol. 28. Melville, NY: American Institute of Physics.

8.4.18 Improving Probability of Detection Assessment for Non-destructive Testing *(C Harding and G Hugo, [DSTO])*

When aeronautical fatigue is managed using nondestructive testing (NDT), knowledge of detectable defect sizes is essential to ensure safe inspection intervals are set. Standards often require that the detectable defect size be demonstrated to have a minimum probability of detection (POD) for the particular inspection technique employed. Unfortunately, there is often a significant gap between what the structural integrity engineers assume has been demonstrated and the actual assessment of NDT reliability that was undertaken [1-3]. Measuring POD is very expensive. Current practices require large-scale trials of NDT procedures on representative components to gather data for statistical analysis.

DSTO has an ongoing research program aimed at reducing the gap between what is assumed and what occurs in practice in the assessment of the reliability of NDT. This gap is being addressed using a variety of approaches including:

- Reducing the cost of conventional POD trials by improving the statistical analysis tools applicable to small data sets [3, 4].
- Developing model-assisted approaches to POD assessment, aimed at reducing the cost of high-fidelity POD assessment [5, 6].
- Developing tools for low-fidelity POD assessment to enable NDT reliability to be explicitly considered in applications for which a full POD trial cannot be justified.
- Improving communication and understanding between the NDT community and the engineering community [7].

A model-assisted probability of detection (MAPOD) assessment was undertaken to determine the POD for an automated ultrasonic inspection for fatigue cracks at fastener holes in the lower wing skin of the RAAF F-111 aircraft [8]. Understanding the reliability issues for this inspection and measuring the POD are important for both airworthiness assurance for the F-111 and facilitating acceptance of a new NDT technology for use on flight-critical structure.

A transfer function modelling approach was employed, which used data for (i) laboratory-grown fatigue cracks in simple specimens, (ii) artificial notches in the complex structure of actual F-111 components, and (iii) the baseline case of artificial notches in simple specimens. The ultrasonic response was modelled for each of these cases and used to estimate the ultrasonic response from genuine fatigue cracks in actual wing structure. A POD trial provided data to quantify the effects of complex structure on the ultrasonic response and human factors in NDT technicians' interpretation of the ultrasonic data. Significant differences in ultrasonic response were observed between artificial notches and fatigue cracks, with fatigue cracks giving generally lower amplitude and area, and also exhibiting a larger variance in the response. For this inspection, the greater variability in response of the fatigue cracks compared to notches is a key factor influencing POD.

The transfer function POD modelling approach is ideal for this type of inspection and provides a strategy to account for both the nature of the defect and the complexity of the structure, as well as incorporating human factors in interpreting the acquired data.

References:
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2. Gallagher, J.P., et al. (2007) "Demonstrating the Effectiveness of an Inspection System to Detect Cracks in Safety of Flight Structure" in *10th DoD-NASA-FAA Aging Aircraft Conference*. Palm Springs, CA.
3. Harding, C.A., (2008) *Methods for Assessment of Probability of Detection for Nondestructive Inspections*, PhD Thesis, The University of Melbourne.
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8.4.19 Probability of Detection for Structural Health Monitoring, *(C Harding and G Hugo, [DSTO])*

Mature processes and standards exist for the validation and certification of non-destructive inspection (NDI) procedures for aircraft structural integrity applications. Reliance on structural health monitoring (SHM) in place of conventional NDI will require a similar level of confidence in system reliability. However, interpretation of NDI validation requirements in the context of SHM is not straight-forward.

One of the foundations of the traditional approach for validating performance of NDI in terms of probability of detection (POD) is that every time a component is inspected, the outcome is independent of any previous inspections. This approach cannot be directly transferred to continuously-monitored systems or permanently-mounted sensors with intermittent interrogation because independence cannot be assumed [1]. A framework for validation of SHM technologies needs to consider the overall risk of missing a defect before it reaches a critical size.

DSTO is undertaking an investigation into the relevance and possible interpretations of existing NDI standards to SHM for structural integrity assurance. The study is focussed on the certification of SHM systems used for the detection and characterisation of discrete defects to meet structural integrity management standards.

References:
1. Hugo, G.R. and C.A. Harding. (2008) "Requirements for NDI Procedure Validation and Implications for SHM Technologies" in *2nd Asia-Pacific Workshop on Structural Health Monitoring*. 2-4 December, Melbourne, Vic, Australia.

8.4.20 Structural Design Standards and Structural Certification of UAVs, *(J Morrish and P Jackson, [DSTO])*

Airworthiness authorities world wide are continuing to develop the design and airworthiness rules for the wide range of UAV configurations coming into service. DSTO has been conducting preliminary investigations into the implications for initial certification and ongoing airframe structural integrity management of the different ways of categorising UAVs and the associated selection and tailoring of appropriate standards. Of particular interest is establishing the relative levels of safety (required by the ADF Airworthiness Management Authorities) obtained from the deterministic

"Code of Requirements" vs the probabilistic "System Probability of Failure" design and verification approaches. The goal is to provide generic guidance to support certification and airworthiness management programs for (largely) composite UAV structures, of traditional and novel configurations and modes of operation. The work will be on-going over the next year.

8.4.21 Alternative Approach for Test Interpretation, *(G Chen [DSTO])*

As part of the structural integrity support for the F-111 aircraft provided in the Sole Operator environment, a certification basis was required for the F/D-model wings fitted to the RAAF fleet. This requirement led to DSTO testing of an ex-USAF wing as the F-111 F-WELD Test. After the application of significant load cycles representing several service lifetimes, the test article was torn down. Two large fractures were identified visually in the Bulkhead 2 area, a type of failure that has yet to be observed in other retired fleet wings. The large cracks initially presented concerns for fleet management due to the difficulty of inspecting the location. Studies were carried out to understand the relevance of cracks to the RAAF operations.

Fractographic analysis performed for the two cracks found that one, designated “Crack 2” in a number of reports, appeared to have originated from a large surface inclusion but was not able to identify the exact causes and rates of crack growth as no regular CPLT marks were found on the fracture surface. Therefore, an alternative approach using finite element and crack growth analyses coupled with analytical stress concentration modelling was explored for test interpretation.

In this approach, a FE stress analysis based on the Internal Loads Model (ILM) was first conducted, which indicated that the flight loading simulated by the actuators during F-WELD testing may be producing higher than expected local bending stresses in the Bulkhead 2 aft bay area. However, an initial crack growth analysis with the maximum stress from ILM FE analysis predicted lives which were far beyond the F-WELD test duration. Since a refinement of the ILM mesh or creation of a fine grid model for Bulkhead 2 was not possible, an alternative method using stress concentration analysis was explored to refine the maximum stress at the crack locations.

From the fractographic analysis, it was found that Crack 2 had initiated at a large surface inclusion, the stress concentration analysis was conducted with and without considering the inclusion. The crack locations without considering the inclusion were modelled as shoulder fillets. The stresses with considering the inclusion were analysed by considering that the cavity was filled with a material having a different modulus of elasticity. Two important assumptions were made: there was no defect in the adhesion between the two materials; and the shape of the inclusion was between spherical and ellipsoid. The inclusion was assumed as a material with a modulus of elasticity, E'. The ratio of the two moduli of elasticity, E'/E, can be assumed to be between 0 and ∞. E represents the modulus of elasticity for the F-111 aluminium wing material (Al 2024-T851). The rigid inclusion assumption was conservatively assumed in this case. The stress concentration factors with and without considering the inclusion were derived.

The derived analytical Stress Concentration Factors refined the maximum stress from FE analysis. With these stresses, further crack growth and spectrum severity analyses were conducted, suggesting that: (1) the inclusion increased the crack growth rate, but its absence would not have prevented cracking, and (2) considerably higher stresses caused by the discrete actuator loading during the test can significantly reduce the fatigue life.

In summary, because the fractographic analysis as a conventional test interpretation activity was not able to identify the exact causes of crack and crack growth rates, an alternative approach was developed for test interpretation. Based on FEA, stress concentrations and crack growth analyses conducted in this approach, the cracking found in Bulkhead 2 was caused by the local loading effects from the test rig actuators generating high local bending and torque loads and would not lead to structural failure of the wing, thus it was concluded that these cracks were not concerns for RAAF F-111 aircraft fleet management. This alternative approach can be applicable to the similar engineering problems.

8.4.22 Application of SIFT to micro-cracking and lug failure *(G Pearce, L Djukic, R Wootton and D Kelly [University of New South Wales in conjunction with the CRC-ACS]).*

A User Subroutine has been developed to implement the Strain Invariant Failure Theory (SIFT) to composite structures in the MSC Marc finite element analysis environment [1]. The algorithm applies a micromechanical modelling procedure to determine strains in the matrix between fibres, at the interface between resin and fibres and in the fibres themselves to predict initiation of failure in composite structures. Applications described in this section include micro-cracking in woven fabrics caused by thermal residual stresses and an investigation of inter-laminar failure in a composite lug subject to bearing loads.

Lugs are used as fittings in aerospace structures to form connections between structural components. Composite materials provide a lightweight design solution but are subject to a wider range of failure modes than fittings made from metals. In addition the lugs are highly loaded and attempts have been made to create fibre architectures in which the fibres are optimally aligned to increase the stiffness and strength. In one such research program an unexpected failure mode was encountered when the thickness of the lug was increased. Strength improvements were obtained for 6mm thick lugs when steered fibres were included in the laminate via a set of tailored fibre pattern (TFP) layers. The improvement was erased in 12mm thick lugs by the appearance of a failure mode that involved splitting of the laminate between the steered layer of the laminate and the neighbouring plies. This failure mode can be seen clearly in Figure 54. This splitting mode was not present in the composite lugs without TFP layers.

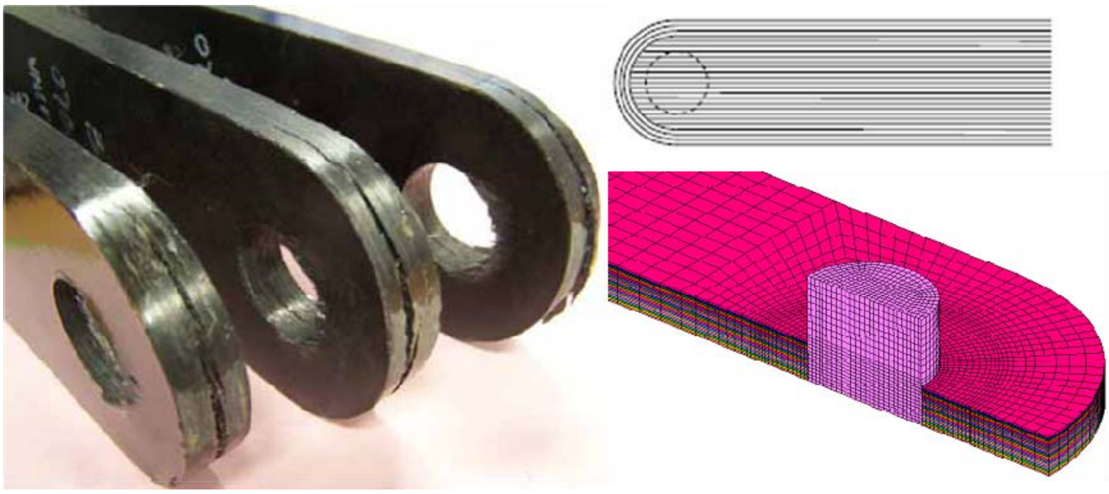

Figure 54 Splitting mode of lug with TFP layers, Fibre orientation in tailored fibre pattern (TFP) and lug Finite Element model (Figure reproduced with permission of CRC-ACS)

The preliminary results of the SIFT analysis are shown in Figure 55. The region shown is the first ply above the central TFP layer. The dark locations show regions of damage predicted by the SIFT subroutine. It can be clearly seen that the drop-off of the axial steered fibres is initiating a new failure mechanism that is separate from the initial bearing damage.

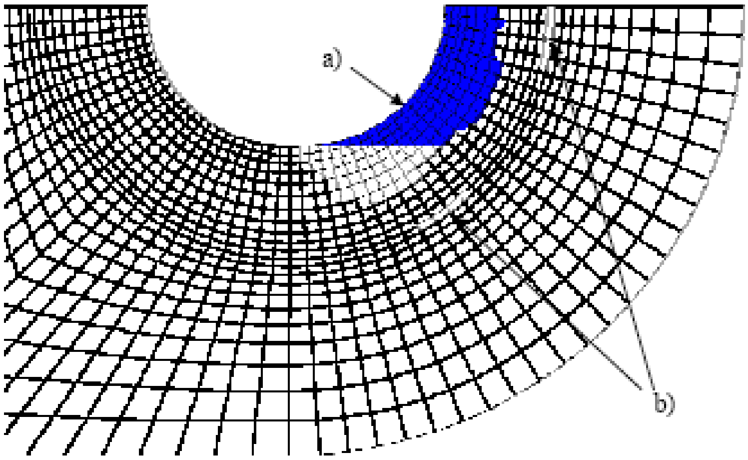

Figure 55 Preliminary SIFT results showing a) Bearing failure region and b) Splitting initiated at axial fibre termination.

The analysis of failure in a composite lug considers progressive failure in an application for which a fracture mechanics approach may be better suited. However the growth of cracks in the resin between fibres in a composite structure is complex and the crack tip can be blunted as it encounters fibres and grows around the circumference of the fibre rather than through a homogeneous resin layer. These observations have led to the application of SIFT to the progressive bearing failure and the results appear to give a qualitative understanding of the initiation of failure at the edge of the fibre steered region for much lower effort than a full fracture mechanics analysis.

1. Application of SIFT to micro-cracking and lug failure G Pearce, L Djukic, R Wootton and D Kelly, Proc AIAC-13 Thirteenth Australian International Aerospace Congress, Melbourne March 2009

8.4.23 Fuzzy Corrosion Fatigue Model for Aging Aircraft Structure Reliability *(Yunxiang Che, He Ren, Xu Wang [RMIT University])*

The integrity and safety of aircraft structure can be greatly degraded because of corrosion and corrosion fatigue. However, the failure mechanisms involved are very complicated and exhibit randomness and fuzziness. This program proposes a fuzzy reliability approach to improve our ability to describe the corrosion fatigue life of aircraft structure, based on the service data obtained for aging aircraft. The effects of the pit aspect ratio, the crack aspect ratio and all fuzzy factors on corrosion fatigue life of aircraft structure are discussed, and used to identify an expression for corrosion fatigue life.

The model was used in a Monte Carlo simulation to assess the effects of the pit aspect ratio, as shown in *Figure 56*.

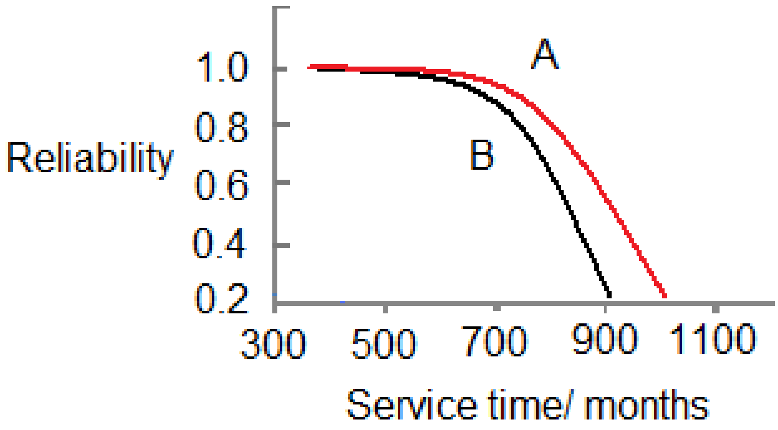

Figure 56 Effects of the aspect ratio on corrosion fatigue life

Not only the effects of random factors but also those of fuzzy factors on corrosion fatigue life can be taken into consideration by using the fuzzy reliability method adopted, and this is considered more reasonable than the conventional reliability approach. In view of the analytical results, an aging structure reliability is significantly affected by the fuzzy threshold stress intensity range parameter ΔK_{th}, which is based on materials testing. Further research on the parameter is needed to analyse the accuracy of membership function. Pit growth is related to engineering practice and to manufacturing technique, and it is important to derive a more concise mechanics-based model to simulate this stage, instead of an empirical assumption. Future analysis may emphasize the electrochemical mechanisms and the factors that have significant effects on corrosion growth.

Reference

1. Fuzzy Corrosion Fatigue Model for Aging Aircraft Structure Reliability, Yunxiang Che, He Ren, Xu Wang, Proc AIAC-13 Thirteenth Australian International Aerospace Congress, Melbourne March 2009

8.4.24 Rapid Operational Loads Measurement (ROLM) of Caribou tailplane using the CMPLE system, *(I Powlesland and Steve Galea, [DSTO]).*

Introduction

DSTO has developed, under sponsorship from RAAF Aircraft Structural Integrity – Director General Technical Airworthiness (ASI-DGTA), a capability for quickly recording the strain levels on an operational aircraft. The system, called the Compact Multi-Parameter Loads Evaluation (CMPLE) system, is fully autonomous, quick and easy to calibrate, mount and remove, platform independent and able to record data from a small number of flights that included the actual manoeuvres of interest. This activity is known as Rapid Operational Loads Measurement (ROLM). Whilst the CMPLE system was under development, ASI-DGTA needed to substantiate the Caribou horizontal tailplane Life-Of-Type (LOT), based on RAAF usage. In this case the ASI-DGTA needed to determine the peak load/stress at the critical tailplane structural locations of Caribou, and CMPLE system was used to measure critical loading action in flight trials on the Caribou horizontal tailplane [1, 2]

System Overview

The CMPLE system consists of 4 main components, viz., (1) sensing, (2) carrier, and (3) interface modules, and (4) support case. The system setup is shown schematically in *Figure 57.* The sensing module contains an internal battery, memory and signal conditioning electronics. This module also contains two strain sensors with their sensitive axis at 90° to longitudinal axis of the module, one temperature sensor, a 3 axis accelerometer and a timer. The two strain gauges were of different types, a semiconductor gauge having higher noise immunity and an electrical-resistance foil gauge being less temperature sensitive. The sensing units have one Gb of flash memory, a maximum sampling rate of 1024 samples/sec, a lithium polymer battery, a maximum strain range of 2000 με and an operating temperature range of -20 to 60°C. Although both strain and acceleration sensors are installed only one, selected by software, may be employed for a given run. A detailed description of the system and its development is given in references [3]

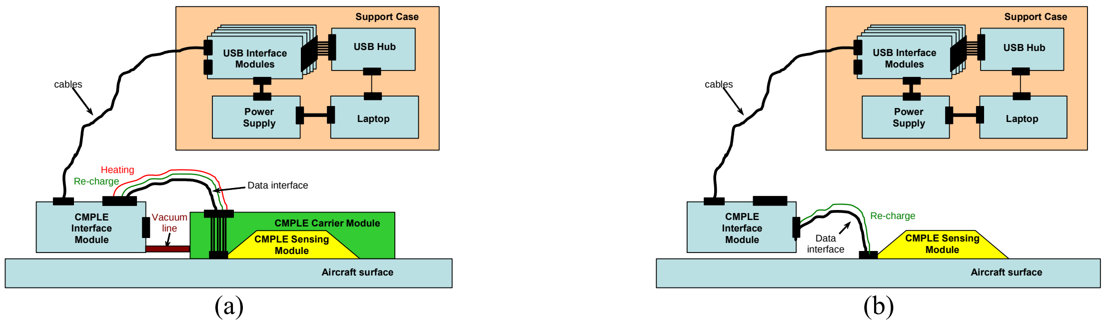

Figure 57. Schematic of the CMPLE system in the (a) installation and (b) pre/post-flight phases.

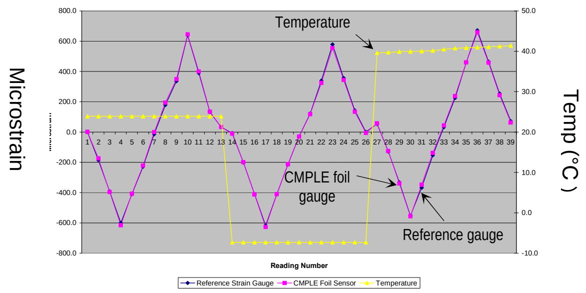

Figure 58. Calibration of CMPLE foil sensor with reference strain gauge under thermo-mechanical loading.

Calibration

Prior to the flight trial each sensing unit was subjected to a thermo-mechanical loading (strain amplitude of ±1500 με over a temperature range of -10°C to 40°C) in the laboratory using a 4-point bending rig to obtain baseline calibration data. Typical calibration curves are shown in *Figure 58*. ElectRelease™ [4 ,5] adhesive was used to attach the sensing units, and allowed the units to be attached and removed to the specimen numerous times without damaging the modules. ElectRelease™ is a family of epoxy resin adhesives that can be easily removed after cure by passing an electrical current through the adhesive.

Loads flight trial

Installation

The carrier was used to attach the sensing module onto the aircraft. For the Caribou flight trial, DSTO produced one carrier module and eight sensing modules. Removal of paint and, installation and removal of the modules and data download was undertaken on Caribou (DHC-4) A4-299 by Australian Aerospace (AA) at their facilities at Brisbane Airport. Typically the installation of the sensing module, after paint was removed, took approximately 60 minutes to perform. In this loads flight trial six sensor modules were fitted to the aircraft, four strain and two acceleration units. Photographs of the installation configuration are shown in *Figure 59(a)* and *(b)*, with the attached sensing units shown in *Figure 59(c)*.

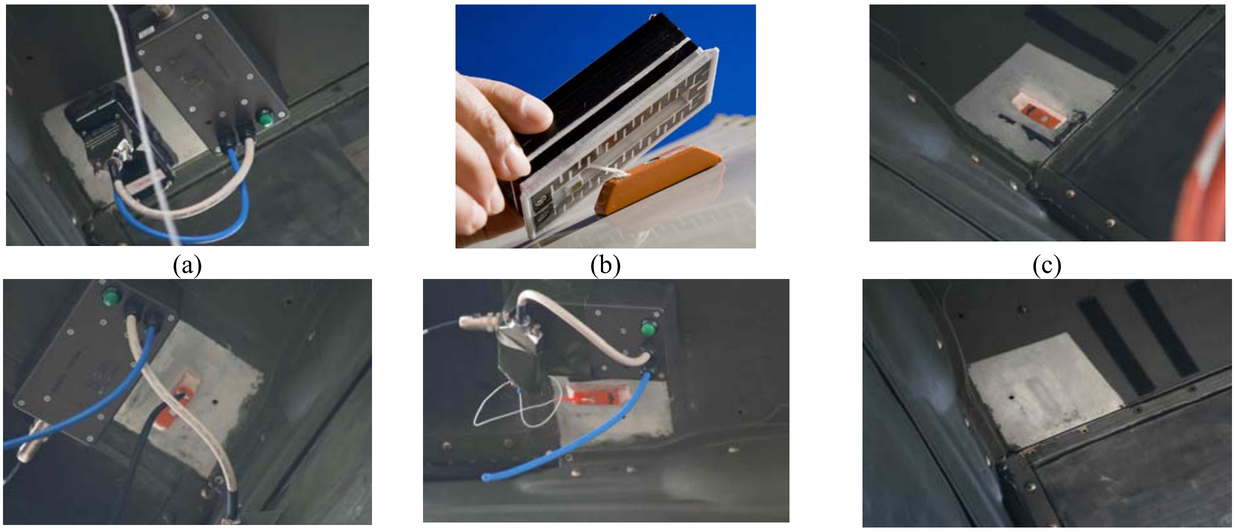

Figure 59. (a) The carrier and interface on aircraft during the attachment of the sensing unit, (b) removing the carrier leaving the sensing unit attached to aircraft skin, (c) the sensing unit attached to the Caribou lower starboard horizontal tailplane, (d) data download, battery re-charge and trigger arming configuration, (e) CMPLE system configuration during the removal phase and (f) the Caribou skin after sensing unit was removed.

Data collection

To enable data download and arming the sensing units were connected to the support case via the interface. The system configuration allowing data download, re-charging the battery and trigger arming is shown in *Figure 59(d)*. All units were armed simultaneously, typically after the flight data has been downloaded from all units, so that they start recording at the same time before the next test flight.

Removal

For removal the sensing modules were connected via the interface to the support case and "Debond" mode enabled. The connection setup, shown in *Figure 59(e),* allowed 48 V to be applied across the ElectRelease adhesive. After applying the voltage for about 45 minutes the units came away easily under light finger pressure leaving a clean surface with very little adhesive residue on the aircraft (see *Figure 59(f)*). In fact the adhesive layer stayed attached to the sensing modules in all cases.

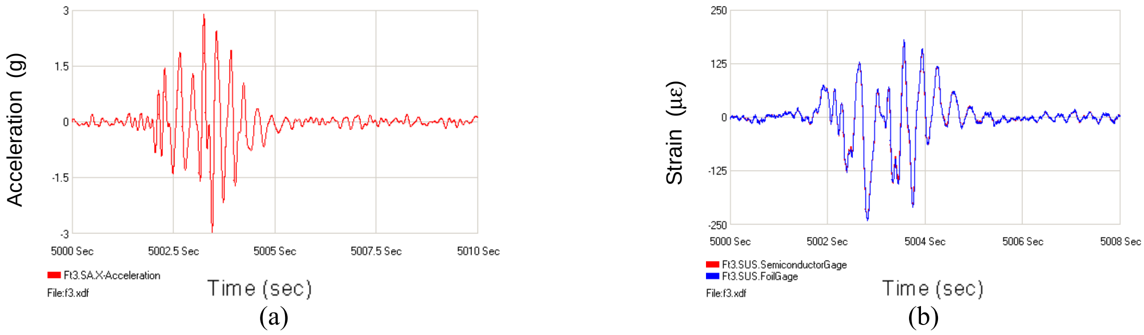

Figure 60. (a) Stbd tailplane tip acceleration and (b) stbd upper leading edge spar root strain time response during stall in Flight 3.

Results

Typical time history response of starboard (stbd) accelerometer and stbd upper leading edge spar root strain for a stall over the 5000 to 5010 sec window of Flight 3 is shown in *Figure 60(a)* and *(b)*, respectively. The resonant frequencies calculated, from the acceleration time history data, are shown in *Figure 61* and correspond reasonable well to the GVT undertaken previously. The internal consistency of the foil and semi-conductor strain sensors on the stbd leading root edge root readings were good (see *Figure 60(b)*). However significant variations between the foil and semi-conductor

sensors were observed for the two port sensing units, see [1]. The time history strain data was then analysed to calculate the mean strain exceedences for the stbd upper leading edge spar root of the tailplane. A summary of the results are plotted in *Figure 62*. Temperature profiles were also recorded and gives a summary of the temperatures measured for the 6 units on Caribou in Flight 3.

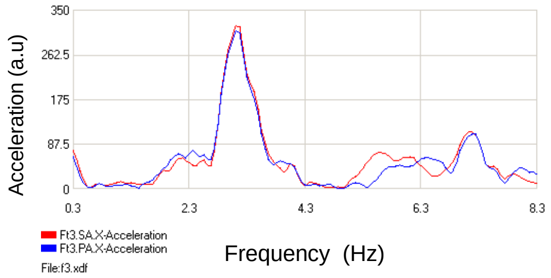

Figure 61. Frequency response of port and stbd accelerometers in Flt 3.

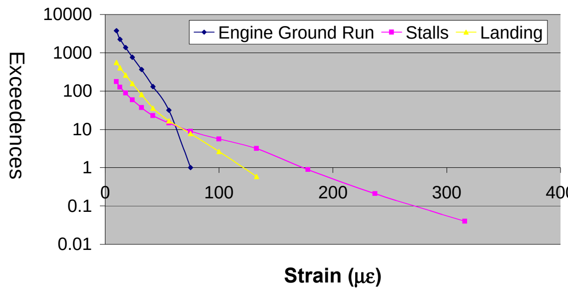

Figure 62. Summary of mean exceedances for the entire flight trial for stbd upper leading edge spar root strain.

Table 1. Temperature extremes recorded during Flight 3

Sensor location	**Max temp (°C)**	**Min temp (°C)**
Stbd upper tailplane root	37.4	3.2
Stbd lower tailplane root	30.2	2.4
Stbd tailplane tip	41.4	2.9
Port upper tailplane root	41.2	2.7
Port lower tailplane root	32.9	2.5
Port tailplane tip	44.3	3.0

Summary

A fully autonomous and platform independent system which is able to record strain and temperature or acceleration and temperature for a small number of flights was developed and demonstrated. The system is quick and easy to calibrate, mount and remove. The sensing units were installed on Caribou in relatively short space of time and were successfully used to measure the flight loads on the horizontal tailplane. All the sensing units survived the three flights and were removed undamaged as planned. The trial collect loads, acceleration and temperature flight data for Caribou during a limited flight trial and successfully demonstrated the practicality of reading strain, acceleration and temperature in flight without the need for any wiring, significant hardware, aircraft power or special mounting brackets.

Future activities

Lessons learned during the Caribou loads flight trial will be used to modify the CMPLE system to make it more robust and versatile. For example, replacing the two unidirectional strain sensors with one rosette strain gauge within the sensing units and to have strain only sensing and acceleration only units. Also the procedure of having to connect cables to sensing units located in inconvenient positions for data download and arming caused extra workload. A wireless link, such as acoustic, infrared (IR), or RF, would simplify the process.

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8.5 FATIGUE INVESTIGATIONS IN NEW ZEALAND

8.5.1 Development of a fleet management tool based on flight-by-flight damage tracking algorithms *(S J Houghton, S K Campbell [RNZAF])*

The RNZAF C-130 Life Extension Project (LEP) currently under way in North America includes substantial refurbishment of the wing structure. Analytical estimates of the magnitude of the fatigue life extension provided by the structural upgrade were based on fatigue damage rates derived from the Statement of Operating Intent (SOI) for the fleet. The need to check the assumptions underpinning the SOI led to a requirement for an assessment of current aircraft usage in service.

The resulting Usage Monitoring System installed in a single C-130 airframe, and reported at previous ICAF Conferences, utilises a commercial digital data recorder and has the capability to track both parametric (Airspeed, Altitude, Pressure, Nz) and Engineering Data (strain at 2 critical locations). The ground support system developed for the trial includes fully automated error detection and data reduction. Fatigue and usage data analysis is based on commercial graphical environment software. The software produces flight by flight mission profile graphs, enables the development of a cumulative Nz exceedance curve, and publishes these to a web based reporting system. This system has been operational for over 2 years and has produced over data for over 1000 flight hours.

Recently the data analysis system was expanded to include a fleet management tool based on flight by flight damage tracking. As a baseline, the strains at two fatigue critical locations on the C-130 centre wing were measured for each of mission profile defined in the RNZAF Statement of Operating Intent (SOI). A cycle by cycle crack growth analysis was conducted for each SOI mission profile. The analyses assumed a standard 0.050” crack at a 3/16 fastener hole, and were conducted using a modified Forman equation for crack growth with no retardation effects. The crack growth increment for these baseline SOI profiles is used to normalise the crack growth increments for operational usage calculated on a flight by flight basis using the same assumed crack geometry. The normalised flight by flight crack growth increments provide a fleet management tool through which fatigue life consumption rates and relative mission severity (measured against SOI) can be tracked on individual aircraft. Future plans included automatic mission profile detection and changes to the crack growth model to reflect actual geometry of each location.

8.5.2 CT-4E Usage Monitoring *(R E Brookes, S K Campbell [RNZAF])*

The Royal New Zealand Air Force (RNZAF) operates a fleet of CT4-E Air Trainers. The CT4-E is an upgrade to the CT4-B, which the RNZAF had previously operated for a number of years. The CT-4E has a higher empty weight and increased engine power over the CT-4B.

Structural management practices for the CT-4E have been based on the Statement of Operating Intent (SOI) and fatigue test data for the CT-4B however a recent review concluded that amended structural life management procedures would be needed for the CT-4E. The RNZAF have instigated a usage monitoring and individual aircraft tracking programme to measure parametric and mission profile data for the CT-4E fleet to ensure that the current fleet management process is appropriate. The programme uses a “commercial off the shelf” integrated sensor and acquisition system (Appero GAU1000).

8.5.3 Contributions to the Development of 3-D Stress Intensity Factor Handbook *(S K Campbell, S J Houghton [RNZAF])*

Handbook solutions to determine the Stress Intensity Factor (SIF) for various engineering problems have existed for a number of years. Invariably these solutions have been produced for idealised structural configurations. An extensive library of SIF solutions for cracks in two dimensional plates exists, and a limited number of 3-D solutions have been published. It is becoming commonplace to develop computational models of 3-D cracking problems and to calculate 3-D SIF’s on a case by case basis. Such an approach is time consuming and can be computationally difficult. As part of a collaborative programme under the Technical Cooperation Programme (TTCP) between the Governments of

Australia, Canada, New Zealand, The United Kingdom and the United States, work has begun on developing a handbook of 3-D SIF's. The handbook will be largely developed through the creation of parametric finite element models of common 3-D structural configurations.

To date, the New Zealand contribution has focused on development of SIF for corner and through cracks in the presence of strong compressive residual stress fields. This work is focused on determining the SIF for a cold worked fastener hole in a structural element with low edge distance.

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19. ABSTRACT

This document has been prepared for presentation to the 31st Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Rotterdam, the Netherlands, 25th and 26 May 2009. Brief summaries and references are provided on the aircraft fatigue research and associated activities of research laboratories, universities, and aerospace companies in Australia and New Zealand during the period April 2007 to March 2009. The review covers fatigue–related research programs as well as fatigue investigations on specific military and civil aircraft.